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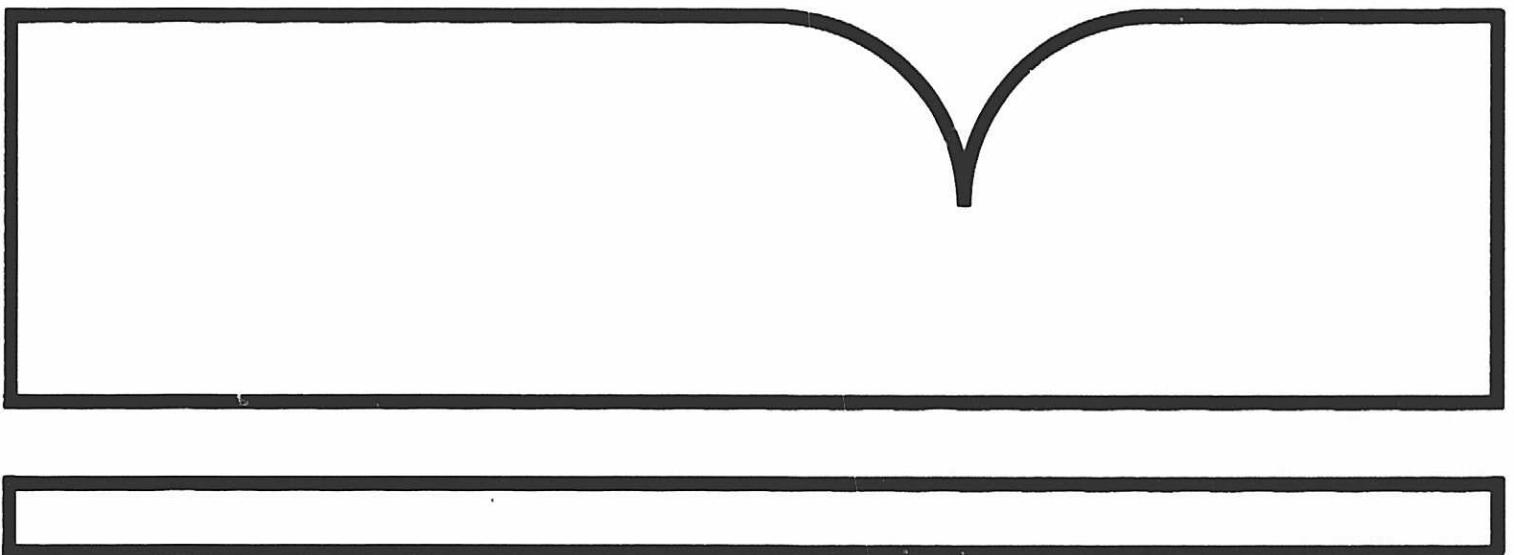
Instrument Interface Description for NOAA 2000 Instruments with
European Morning Spacecraft and/or NOAA O, P and Q Spacecraft

(U.S.) National Aeronautics and Space Administration, Greenbelt, MD

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The purpose of this document is to describe at a high level the common interface provisions and constraints placed on the NOAA-2000 instruments and the interfacing spacecraft elements in the following areas: electrical interface, mechanical interface, thermal interface, magnetic interface, electromagnetic compatibility, structural/mechanical environmental interface, contamination control, and the ionizing radiation environment. The requirements reflect the fact that these instruments must be compatible with a number of different polar orbiting satellite vehicles including the NOAA-OPQ satellites and the EUMETSAT METOP satellites. [It is recognized that a number of the detailed interface parameters are products of interactive processes and will evolve during the development phases of the program. These interfaces will be documented in a series of instrument specific "Unique Instrument Interface Descriptions".]

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for
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with
European Morning Spacecraft
and/or
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1.0 Scope

The purpose of this document is to describe at a high level the common interface provisions and constraints placed on the NOAA-2000 instruments and the interfacing spacecraft elements. The requirements reflect the fact that these instruments must be compatible with a number of different polar orbiting satellite vehicles including the NOAA O, P & Q satellites and the European Morning satellites. It is recognized that a number of the detailed interface parameters are products of interactive processes and will evolve during the developmental phases of the program.

2.0 Referenced Documents.

<u>Document Number</u>	<u>Document Title</u>	<u>Using Paragraph</u>
NASA RP 1124	Outgassing Data for Selecting Outgassing Materials	3.7
Mil-Std-454J 1 Mar 1985	Standard General Requirements for Electronic Equipment	3.7 a
Fed-Std-209B	Clean Room and Work Station Requirements, Controlled Environment	3.7 b
Mil-Std-1553b 21 Sep 1978	Aircraft Internal Time Division Command/Response Multiplex Data Bus	3.1.5.1
PPL-19 October 1989	GSFC Preferred Parts List PPL-19	3.1.3
CCSDS 301.0-B-2 Issue 2 April 1990	Recommendation for Space Data System Standards for Time Code Formats	3.1.6 c,
NASA x-600-87-11	Metsat Charged Particle Environment Study	3.8
MIL-M-38510/104B 29 May 1987	Military Specification, Microcircuits, Linear Line Drivers and Receivers, Monolithic Silicon	3.1.5.3, 3.1.5.4
Mil-Std-461C	Electromagnetic Emission and Susceptibility Requirements for Control of Electromagnetic Interference	3.5.1.2, 3.5.2, 3.5.3
Mil-Std-462	Electromagnetic Emissions and Susceptibility, Test Methods for	3.5.1.2, 3.5.2, 3.5.3

Mil-Std-463	Definitions and Systems of Units, Electromagnetic Interference and Electromagnetic Compatibility	3.5.1.2
Mil-B-5087 31 Aug 1970	Bonding, Electric and Lightning Protection for Aerospace Systems	3.1.2
GSFC-S-480-72	Command/Telemetry Bus, General Specification for	3.1.5.1.1, 3.1.6
GSFC-S-480-73	Low Rate Data Bus, General Specification for	3.1.5.1.2
GSFC-S-480-74	High Rate Data Link, General Specification for	3.1.5.2
GSFC-S-311-P-18-03	Thermistor, Specification for	3.1.5.5 a
CCSDS 501.0-B-1	Radio Metric and Orbit Data	3.1.7 c, 3.1.8

3.0 Interface Descriptions.

3.1 Electrical Interface Specification

The Instrument and the spacecraft share electrical interfaces to facilitate the transfer of power, commands and data. Figure 1 depicts schematically the electrical interface. The instruments shall provide independent, totally redundant electrical interfaces except in the cases of the Main Power Line (See 3.1.1.1.1), the Pulsed Load Power Line (see 3.1.1.1.2), the survival heater power line and the survival temperature telemetry sensors. The requirement for and implementation of circuit or functional redundancy internal to the instruments is dependant on the instrument unique scheme for achieving specified reliability and life. Detailed interface requirements are described herein.

3.1.1 Power

The spacecraft shall supply instruments +28 Volt Direct Current power on three switched circuits. Two of the circuits shall provide regulated, conditioned power. These are the Main Power Circuit and the Pulse Load Power Circuit. The third circuit, the Survival Heater Power Circuit, shall provide unregulated nominal +28 Volt DC power for use by the instrument survival heaters. The spacecraft shall provide overcurrent sensing and protection on these power circuits. The instrument shall not provide fuzes on any power circuit. The instrument shall not be damaged by the simultaneous or by any sequential application or removal of power from the power circuits. The instrument shall operate within its specified performance limits when powered as specified herein. Instrument time-domain power consumption profiles addressing peak and average power in all modes shall be determined for each power circuit, and the values shall be documented in the instrument unique interface control document. Spacecraft power circuit characteristics and instrument load characteristics shall comply with the limits specified herein.

3.1.1.1 Regulated Power Characteristics

The Spacecraft shall supply to the instruments regulated nominal +28 Volt Direct Current power on the Main Power circuit and the Pulsed Load Power circuit.

Main and pulsed load power are applied to the instrument at turn-on and when the instrument is in an operational mode. Main and pulsed load power are not supplied during launch or orbital insertion operations.

The main power circuit provides well regulated power with minimum load-induced transients and ripple and is intended to be the principle source of steady state power for the instrument. This supply has limited tolerance for instrument load dynamics as specified in paragraph 3.1.1.3.1 below.

The pulse load power circuit supplies well regulated power with greater tolerance for load-induced ripple and transients. It is intended to power instrument loads such as stepper motors, operational heaters, and other circuits which by their nature cannot meet the ripple and transient limits that are imposed on loads which are powered by the main power circuit.

Main and pulsed load power, once applied, may normally be removed from the instrument at any time after the instrument has been configured to a SAFE condition. The instrument shall, however, not be damaged by the removal of both or either the main or pulsed load power while the instrument is in any operating state.

The spacecraft provides redundant main and pulsed load power sources. Instruments having redundant circuits powered by the main power circuits shall provide redundant main power connections. Instruments that do not have redundant circuits powered by the main power circuits may elect to have a single main power connection, in which case the spacecraft shall provide the necessary redundant source switching, utilizing a single main power line. The instrument main power circuit terminal configuration shall be described in the instrument unique interface control document.

Instruments having redundant circuits powered by the Pulse Load power circuits shall provide redundant pulse load power circuit connections. Instruments that do not have redundant circuits powered by the pulse load power circuits may elect to have a single pulse load power connection, in which case the spacecraft shall provide the necessary redundant source switching, utilizing a single pulsed load power line. The instrument pulse load power circuit terminal configuration shall be described in the instrument unique interface control document.

A. Normal Operating Limits - The instrument shall meet all performance requirements when supplied operating power within the normal operating limits specified in this document. The normal operating limits applicable to the main and pulsed load Power Circuits are defined herein.

1. Source Regulation - The voltage, exclusive of ripple and transients, on the main and pulsed load power circuits shall be, when applied, $+28 \pm 0.56$ Volts with respect to the circuit's power return as measured at the spacecraft power source.

2. Source Voltage Ripple - Steady state voltage ripple including repetitive spikes, superimposed on the line voltage, as measured at the spacecraft power source under specified instrument loads, shall not exceed 200 millivolts peak to peak inclusive of all frequency components below 100 kHz.

3. Source Transient Performance (Operational). The instrument shall be exposed to source transients from multiple sources. Source transients may be additive but shall not cause the source voltage to drop below +16 volts nor exceed +38 volts.

a. Low frequency load induced transients - Voltage transients may occur, at the main power source, simultaneously with the application or removal of power, or the initiation of mode changes within a load. Voltage transients may occur at the pulsed load power source due to the operation of stepped or pulsed loads (eg. scanner motors, operational heaters, etc.). The load changes may be induced by the instrument, other instruments or spacecraft subsystems.

The spacecraft shall provide main and pulsed load power sources which shall respond to step changes in their respective load currents in accordance with the following. The induced transient voltage, measured at the power source, as superimposed on the steady state voltage, shall comply with the following limits:

i. Waveform - The induced transient voltage waveform shall be a damped sinusoid having a Q at the resonant frequency of no greater than 3.

ii. Sensitivity - The absolute value of the peak induced transient in Volts shall be no greater than 0.133 times the value of the step current value in Amperes.

iii. The maximum rate of change of the induced transient voltage shall not exceed ± 300 volts per second per load Ampere.

iv. The absolute value of peak transient voltage after 60 milliseconds from its initiation shall not exceed 0.05 times its peak value.

b. Random occurrences of positive or negative high frequency voltage transients having integrated values not exceeding 120 mV·s and having amplitudes not exceeding ± 0.75 Volt peak may be present on the main and pulsed load power circuit at the power source.

c. Voltage excursions at the instrument terminals induced by changes in load, effected by the instrument, will reflect the characteristics of the hookup and wiring between the instrument and the power source. The spacecraft shall provide hookup and wiring with the following impedance characteristics between the instrument power terminals and the power source. For frequencies below 1 MHz, the hookup and powerline impedance for each power line wire and each return wire may be represented by a resistor (R1) in series with an Inductor (L1), with this series combination shunted by a resistor (R2). The limiting values of the components are ≤ 0.3 ohms for R1, ≤ 5 microhenries for L1, and ≤ 30 ohms for R2.

d. Random occurrences of positive or negative high frequency voltage transients not exceeding $120 \mu\text{V}\cdot\text{s}$ and having amplitudes not exceeding +10 or -12 volts peak may be present on the main and pulsed load power circuit at the instrument power terminals.

e. Isolation of Main Power Source - The sensitivity of the transient response of the main power source to dynamic loads occurring on the pulsed load power circuits shall be less than 0.1 times its sensitivity to dynamic loads occurring on the main power circuits.

B. Fault Limit Voltages - The instrument shall not be damaged by exposure to the fault voltage limits defined herein.

1. Source Switch-over Fault Limits - Voltage excursions, occurring at the spacecraft power source, leading to or caused by the switch over of spacecraft redundant power sources shall not exceed the limits specified below:

a. Undervoltage Condition - Under voltage fault limits for the main power circuit are defined as voltage excursions below +27 volts but not lower than +16 volts. Under voltage fault limits for the pulsed load power circuit are defined as voltage excursions below +27 volts but not lower than +15 volts. The undervoltage condition may last for periods up to 3 seconds duration.

b. Overvoltage Condition - Over voltage fault limits are defined as voltage excursions above +29.5 volts but not higher than +38 volts. Overvoltage conditions exceeding 29.5 volts shall have a maximum duration determined by the formula:

$$t = 50 \cdot (V - 29.5)^{-1}, \text{ where: } t = \text{time in milliseconds, and, } V \text{ is the level of the line voltage during the overvoltage condition.}$$

Over voltage conditions, once detected, are followed immediately by an undervoltage condition not exceeding 1.5 seconds duration.

The instrument need not meet all of its performance requirements while the source switch-over fault under-voltage or over-voltage condition exists, but it shall autonomously resume operation in the mode it was operating prior to the fault event not later than the second synchronization pulse signal occurring after the correction of the fault condition.

2. Test-Fault Limits - Application to the instrument main and pulsed load power circuit terminals of voltage levels that are within the range defined by 0 to +38 Volts for periods of unlimited duration shall not cause damage to the instrument.

The instrument need not meet all of its performance requirements while the source voltage is out of normal operating limits, but it shall autonomously resume operation in the state it was operating prior to the fault event, or shall autonomously enter a state from which it can be commanded to resume its prior operating state, not later than the second synchronization pulse signal occurring after correction of the fault condition.

3.1.1.2 Unregulated Power Characteristics

The spacecraft shall supply to the instruments unregulated nominal +28 Volt direct current power on the Survival Heater Power Circuit. Survival heater power is applied to the instrument whenever the main power and the pulse load power are not applied. The sole purpose of this source is to supply power to instrument activated survival heaters.

Abnormal application of survival heater power while the instrument is in any operating state shall not, of itself, cause damage to the instrument.

A. Normal Operating Limits - The survival heater power circuit shall provide a direct current voltage, exclusive of ripple and transients, of not less than +22.0 volts dc nor more than +37.0 volts dc with respect to the survival heater power circuit return as measured at the spacecraft power source.

B. Source Impedance - The spacecraft shall provide hookup and wiring with the following impedance characteristics (including up to 0.2 ohms for fuzes) between the instrument power terminals and the power source. For frequencies below 1 MHz, the hookup and powerline impedance for each power line wire and each return wire may be represented by a resistor (R1) in series with an Inductor (L1), with the series combination shunted by a resistor (R2). The limiting values of the components are ≤ 3 ohms for R1, ≤ 5 microhenries for L1, and ≤ 50 ohms for R2.

C. Voltage Variation - The variation of the steady state voltage on the survival heater power circuit measured at the power terminals of the instrument shall not exceed the following limits:

1400 mv peak to peak ripple as detected in a 50 MHz bandwidth; and,

± 1.5 volts spikes, pulse durations from 20 ns to 1μ s, with a prf up to 132 pps.

These voltage variations are defined as having an average value of zero volts as detected in a bandwidth of 10 Hz or less.

D. Fault Condition Limits. Under fault conditions, the bus voltage shall not exceed 42 volts or fall below 0 volts. Recovery to within normal operating limits shall occur within 3 second of the initiation of the fault condition.

Excursions of the power busses outside normal operating limits, but within the fault condition limits described above, shall not result in damage to the instrument survival heater circuits nor preclude their returning to normal operation upon restoration of power to within normal operating limits.

E. Source Voltage Transients (Operational) - Source developed transients super-imposed on the dc level of voltage shall not exceed plus or minus 56 volts peak as measured at the input power terminals of the instrument. The time integral of any source transient shall not exceed three hundred microvolt-seconds.

Switched Load induced source transients resulting in voltage changes of up to $\pm 5V$ lasting for up to 400 μs , and changes of up to $\pm 2 V$ lasting up to 3 ms shall be considered to be within normal limits.

F. Instrument Load induced Transients - Transient voltages induced on the power bus by instrument survival heater circuit operation as measured at the power terminals of the instrument shall not exceed 20 volts peak.

3.1.1.3 Instrument Load Characteristics

Instrument load characteristics limits are defined as measured at the instrument input terminals. Steady State characteristics are defined as those appurtenant to the instrument operating states which have intended durations in excess of 100 milliseconds. Transients characteristics are defined as those appurtenant to any non-repetitive operating events having intended durations of less than 100 milliseconds. Inrush current limiting circuits and/or reactive filtering shall be incorporated, if required, to limit load current rate of change to limits specified herein.

3.1.1.3.1 Load Characteristics, Main Power Circuit

A. Load Current Ripple - The peak to peak amplitude of the steady state load current ripple due to operation in a mode shall not exceed 2% of the maximum average steady state current supplied by the main power circuit to the instrument in that mode. The maximum instantaneous rate of change of the load current shall not exceed 20 $mA \cdot \mu s^{-1}$.

B. Transient Load Current - Exclusive of instrument turn-on, transient load currents drawn by the instrument, for each operating mode, shall not exceed $\pm 50\%$ of the maximum average steady state current supplied by the main power circuit for that mode. Current rate of change limits defined in paragraph 3.1.1.3.1 A apply. All load transients shall be settled, ie. steady state operation attained, within 30 ms from the initiation of the transient.

C. Instrument Turn-On - At instrument Turn-on, the combined transient charge drawn from the Main Power Circuit and the Pulse Load Power Circuit, exclusive of the steady state current, shall

not exceed 40 A·ms with the peak instantaneous current not to exceed 5 A. The current rate of change shall be controlled to not exceed the limits of paragraph 3.1.1.3.1 A for the main power circuit and paragraph 3.1.1.3.2 A for the pulse load power circuit. Steady state load levels shall be achieved within 1 second of the initiation of the turn-on transient.

3.1.1.3.2 Load characteristics, Pulse Load Power Circuit

A. Load Current Ripple - The peak to peak amplitude of the steady state load current ripple due to operation in a mode, exclusive of pulse loads (eg. stepper motor loads) developed by the instrument, shall not exceed 10% of the maximum average steady state current supplied by the pulse load power circuit to the instrument for that mode. The maximum instantaneous rate of change of current shall not exceed $30 \text{ mA} \cdot \mu\text{s}^{-1}$.

B. Maximum Current Pulse Load - The maximum current pulse applied to the pulse load power circuit shall not exceed 1.0 A·s with a maximum amplitude not exceeding and a maximum duration not exceeding 1 second. The maximum instantaneous rate of change of any current pulse applied to the pulse load power circuit shall not exceed $30 \text{ mA} \cdot \mu\text{s}^{-1}$.

3.1.1.3.3 Load Characteristics, Survival Heater Power Circuit.

The maximum instantaneous rate of change of current applied by an instrument load to the survival heater bus shall not exceed $2 \text{ A} \cdot \mu\text{s}^{-1}$.

3.1.2 Grounds

a. Single Point Ground. The spacecraft adheres to a Single Point Ground (SPG) scheme in which no power or signal currents are conducted by the spacecraft structure. This is a means of minimizing ground currents and the effects of ground loops on the performance of the instruments. All connections to the SPG are implemented by the spacecraft. The SPG provides static ground potential to all circuits connected to it. Wires connected to the SPG are not to be utilized by any instrument to carry power or signal return currents. All instrument primary power and signal returns shall be isolated from the instrument chassis and the spacecraft structure. High frequency signals utilizing coaxial or waveguide connections are exempt from this requirement, however, static isolation of the signal from the structure shall be restored as early as is practicable in the signal path (by use of transformers or some other suitable means). Figure 2 is a representation of the SPG grounding implementation.

b. Power Returns. The primary side of each power circuit shall be isolated from structure and all other electrical circuits in the instrument by a minimum of 10 megohms resistance shunted by not more than 0.01 microfarads capacitance. Each primary power return shall be unique and isolated from all other

returns and grounds in the spacecraft. Continuity to the SPG shall be provided by the spacecraft at the power source.

c. Signal Ground. The secondaries of the instrument DC to DC converters shall have their returns referenced to the signal ground and connected at a common internal physical point to the instrument chassis. The signal ground shall be available at a pin on all signal connectors. Signals conveyed through the spacecraft on shielded wire shall have their shields connected, by means of a connector pin designated for that purpose, to the signal ground of the originating instrument assembly.

d. Chassis ground. The chassis of each instrument shall be grounded to the spacecraft through its mounting surfaces. Electrical bonding shall comply with Mil-B-5087, Class R, with a bond resistance not exceeding 2.5 milliohms. If no appropriate bonding surface is available then a grounding strap providing equivalent bond resistance shall be provided. In addition the instrument internal chassis single point ground point shall be connected by as short a wire as possible to a pin in the power connector via which it will be conveyed to the spacecraft single point ground. All hinged items shall be bridged with bonding straps to assure chassis ground integrity across joints. Provisions shall be made to provide instrument chassis ground to all mating connector shells. Signals conveyed through the spacecraft on doubly shielded wire shall have their external shields connected, by means of connector pins designated for that purpose, to chassis ground at both ends of the wire and at every intervening interconnect. Spacecraft harness gross shields shall be connected by means of the connector backshells to chassis ground at all terminations and all intervening interconnects.

3.1.3 Instrument Connectors

a. General. Instruments shall use connectors to interface with the spacecraft harness. Connectors shall be selected in accordance with the GSFC Preferred Parts List PPL-12 and shall be limited to those that accept #20 AWG wire. Round connectors shall be used in applications that interface with spacecraft supplied hardware. Instrument connectors shall provide grounding to the shells of mating connectors. Connectors shall be described in the instrument unique interface control document.

b. Power. Power inputs and their respective returns shall be on the same connector of the instrument. The power/power return connector shall utilize pins. Each circuit shall be conveyed on a minimum of two pins wired in parallel. A pin designated "chassis ground" which is internally connected to the instrument housing shall be provided.

c. Signals and Data. Signals and data shall be on connector(s) other than those provided for power. These connectors shall be female connectors. Each signal/data connector shall provide a connection designated "chassis ground" which is internally connected to the instrument housing chassis grounding point

for the purpose of servicing spacecraft harness external shields. Each signal/data connector shall provide a connection designated "shield ground" which is internally connected to the signal ground for servicing shielded wires within the spacecraft harness. Each signal/data connector shall provide a pin designated "signal ground" which is internally connected to the signal ground for providing reference levels to external electronic equipments. Bi-Polar (two wire) signals shall use connections which are located adjacent to each other. Connections which convey shields shall be located adjacent to the connections which convey the signals which require the shields.

d. Intra-instrument Signals. When an instrument is comprised of more than one cased assembly then intra-instrument signals shall be conveyed by connectors reserved for that purpose.

e. GSE Access Connectors. If an instrument must be accessed by GSE for test purposes while installed in the spacecraft, connectors reserved for that purpose shall be provided.

3.1.4 Test Points

a. Accessibility. Access to test points on the instrument is not normally provided once an instrument is installed in the spacecraft. If access is required, provisions to facilitate it shall be addressed in the instrument unique interface control document.

b. Connectors. Test points other than those accessed through the spacecraft umbilical connector shall be accessed via connectors dedicated for that purpose.

c. Protection. Test points shall be buffered internal to the instrument to prevent damage to the instrument in the event that a test point is shorted to any ground, signal or power circuit.

d. Test point Signal characteristics. Test points shall comply with the following guidelines.

Signal Type: Same as the signal being monitored

Signal amplitude: Representative of the signal being monitored

Output Impedance: 10 kilohms Maximum

Configuration: representative of signal being monitored (i.e. Single ended, referenced to signal ground, differential pair, coaxial, etc.)

Signal ground: Present on at least one pin of the connector

3.1.5 Standard Interface Circuits

Standard interface circuits shall be used. These are defined for all applications of serial digital, digital discrete, and timing signals.

3.1.5.1 Command/Response Multiplex Data Bus

This spacecraft bus conforms to the requirements and protocols of MIL-STD-1553B. Requirements, herein, are described in terms defined in the referenced specification. The command/response multiplex data bus provides two way asynchronous communications service between the instrument and the spacecraft. The instrument shall be configured with an embedded remote terminal. The Spacecraft system is configured as a bus controller. The bus implementations and operations are application specific and each shall comply with the requirements defined in the appropriate sections below.

3.1.5.1.1 Command/Telemetry Bus

Implementation and operation of the Command/Telemetry Bus shall be in conformance with the requirements of specification GSFC-S-480-72, Command/Telemetry Bus, General Specification for. Application of this bus is addressed herein, in paragraphs 3.1.7 c, Serial Digital Commands/Memory Loads, and, paragraph 3.1.9, Housekeeping Telemetry.

3.1.5.1.2 Low-Rate Data Bus

Implementation and operation of the Low-Rate Data Bus shall be in conformance with the requirements of specification GSFC-S-480-73, Low-Rate Data Bus, General Specification for. Application of this bus is addressed herein, in paragraph 3.1.8.1 Low-Rate Science Data.

3.1.5.2 One-Way Data Link

The instrument port shall be configured as a send-Only data transmission terminal. The spacecraft is configured as a receive-only terminal. Communication is asynchronous and is controlled by the spacecraft. The One-Way Data Link shall operate at a burst rate of up to five megabits per second.

Characteristics of, and implementation and operation of the one-way data link shall be in conformance with the requirements of specification GSFC-S-480-74, High-Rate Data Link, General Specification for. Application of this link is addressed herein, in paragraph 3.1.8.2 High-Rate Science Data.

3.1.5.3 Bi-Polar Discrete Circuit

The Bi-Polar discrete circuit utilizes the (generic type) 9614-/9615 line driver/receiver series of terminal devices. The line driver shall conform to the electrical requirements of M38510-/10403. The line receiver shall conform to the electrical requirements of M38510/10404. Component selection criteria shall conform to the requirements of the Instrument development specification.

Figure 3 schematically depicts the implementation of the bi-polar discrete signal.

3.1.5.4 Bi-Polar Clock

The Bi-Polar Clock Circuit utilizes the (generic type) 9614/9615 line driver/receiver series of terminal devices. The line driver shall conform to the electrical requirements of M38510/10403. The line receiver shall conform to the electrical requirements of M38510/10404. Component selection criteria shall conform to the requirements of the Instrument development specification.

Figure 3 schematically depicts the implementation of the bi-polar clock signal.

3.1.5.5 Survival Thermistor Circuit

Each thermistor circuit incorporates at least one and up to three thermistors located in temperature critical areas of the instrument. These thermistors are read directly and periodically by the spacecraft telemetry system when the main power is not applied to the instrument. These thermistors will be biased by a current source, located external to the instrument, while being read.

a. Instrument provisions. The thermistor circuit consists of at least one and no more than three thermistor devices which use a common current return and are accessed via connector by the spacecraft. The thermistor circuit shall be electrically isolated from all other circuits in the instrument. The thermistor current return, likewise isolated from all other instrument circuitry, shall be provided to the spacecraft to facilitate the biasing and reading of the thermistor(s). The thermistor type used shall conform to the requirements of Specification GSFC S311-P-18-03. The exact component selected for use shall be identified by part number in the instrument unique interface control document.

b. Spacecraft provisions. The spacecraft shall provide a current source which generates a current pulse with a peak magnitude of two milliamperes, a maximum duration of one millisecond and a maximum rate of change of current of 100 milliamperes per microsecond applied to each thermistor at times when that thermistor is being read.

3.1.6 Clocks, Synchronization signals and Time

The clock and synchronization signals are supplied to the instruments by the spacecraft. Spacecraft time is provided to the instrument by the spacecraft.

The instruments shall synchronize their operational epochs to the Synchronization Pulse signal described in "b" below.

The instrument shall not be damaged by the absence of the Master Clock signal and/or the Synchronization Pulse signal.

a. Master Clock. A clock signal shall be supplied to the instrument by the spacecraft. Characteristics of the Master Clock signal are described below:

Signal type: constant frequency square wave
Frequency: 1048576 ± 10 Hertz
Short term stability (1 minute): $< \pm 5$ parts per 10^9
Frequency Drift (1 week): $< \pm 3$ parts per 10^8
(1 year): $< \pm 2$ parts per 10^6
Duty Cycle: 50 ± 5 percent of period
Critical timing transition: OFF to ON logic status
Transition Time: TBD

Electrical characteristics of the interface shall be in accordance with the bipolar clock signal defined in paragraph 3.1.5.4.

b. Synchronization Pulse. A logic ON pulse with a repetition period of 8 seconds and a duration of one millisecond shall be supplied to the instrument. This pulse will be derived from the same clock as the Master Clock signal described in "a." above and shall have the same stability and drift characteristics. Its critical timing transition is ON to OFF logic status.

Electrical characteristics of the interface shall be in accordance with the bipolar clock signal defined in paragraph 3.1.5.4.

c. Spacecraft time. Time is provided to the instrument via the Command/telemetry bus. Time is encoded in accordance with CCSDS Recommendation 301.0-B-2, Time Code Formats, using the CCSDS Unsegmented Time Code. Indicated time is that which occurs coincident with the critical timing transition of the next Synchronization Pulse signal. The time message shall be accurate to within ± 1.0 millisecond relative to Greenwich Mean Time (GMT). The least significant bit of the time message shall have a value of 0.1 millisecond or less. Implementation of the time code message shall be in accordance with the requirements of GSFC-S-480-72, Command/Telemetry Bus, General Specification for.

3.1.7 Commands

Instruments are controlled by commands issued from the spacecraft systems. Figure 4 represents a conceptual instrument state diagram about which the command architecture is constructed. Instruments will be supplied commands of the following types and formats.

a. Main power bus. Application of main power shall place the instrument in its Power-On state. The instrument shall be configured to communicate with the spacecraft via the Command-/Telemetry Bus within 100 milliseconds of application of power.

b. SAFE Command. The SAFE command is issued by the spacecraft twenty seconds prior to an anticipated power down. Upon receipt of the SAFE command, the instrument, if it is in any unsafe state, shall transition to the SAFE state. The SAFE state exits automatically to the POWER-ON state upon completion of mechanical and electrical safing procedures.

The SAFE Command is normally in the logic OFF state. The SAFE Command is a logic ON pulse with a duration of one millisecond. The critical transition upon which the instrument shall execute the SAFE command is the ON to OFF logic status transition.

Electrical characteristics of the interface shall be in accordance with the standard bipolar discrete defined in paragraph 3.1.5.3.

The instrument shall be capable of disabling its responsivity to future SAFE Commands by serial command issued via the Command-/Telemetry Bus. The instrument shall, likewise, be capable of restoring this function by serial command.

c. Serial Digital Commands/Memory Loads. All operational commands shall be transmitted from the spacecraft to the instrument via the Command/Telemetry bus. Such commands may call up operational states, request instrument status, request housekeeping data, or perform uploads of spacecraft ancillary information, time or other data (if applicable).

Instrument commands shall be packetized in a format compatible with the requirements for a Version 1 CCSDS packet.

The instrument shall not require that operational state commands be periodically refreshed, ie. the instrument shall maintain its operational status-quo in the absence of periodic serial digital commands.

Instrument specific command requirements and formatting details shall be addressed in the instrument unique interface control document. Any spacecraft provided ancillary data (radio metric and orbit data) shall comply with the requirements of CCSDS 501.0-B-1, Radio Metric and Orbit Data.

Characteristics of the Command/telemetry bus shall be in accordance with the requirements defined for the Command/Response Multiplex Data Bus defined in Paragraph 3.1.5.1. Implementation and operation of the bus shall be in accordance with the requirements of paragraph 3.1.5.1.1.

This bus also provides transmission service for the instrument housekeeping telemetry as specified in paragraph 3.1.9 Housekeeping Telemetry.

3.1.8 Instrument Payload Science data

Instrument payload science data shall be packetized in a format compatible with the requirements for a Version 1 CCSDS packet.

Spacecraft time shall be incorporated into each packet of the payload science data as the time of the occurrence of the first measurement of a data unit within the applicable measurement cycle or epoch.

Instrument Housekeeping Telemetry shall be replicated and incorporated into the payload science data where required for interpretation of the science data.

Payload Science data content, scale factors and formats, packetizing and segmentation specifics shall be addressed in the instrument unique interface control document. However, any spacecraft ancillary data (radio metric and orbit data) which is incorporated therein, shall comply with the requirements of CCSDS 501.0-B-1, Radio Metric and Orbit Data.

3.1.8.1 Low Rate Science Data (< 100 kilobits per second)

Instrument low rate science data is defined as that having a net bit rate (including packetizing overhead) of less than 100 kilobits per second. All low rate science data shall be transmitted from the instrument to the spacecraft via the Low-Rate Data Bus.

Characteristics of the bus shall be in accordance with the requirements defined for the Command/Response Multiplex Data Bus defined in Paragraph 3.1.5.1. Implementation and operation shall be in accordance with the requirements of paragraph 3.1.5.1.2.

3.1.8.2 High Rate Science Data ($\geq$ 100 kilobits per second)

Instrument high rate data is defined as that having a net bit rate (including packetizing overhead) of 100 kilobits per second or greater. All high rate science data shall be transmitted from the instrument to the spacecraft via a dedicated High-Rate Data Link.

Characteristics of, and implementation and operation of the High-Rate Data Link shall be in accordance with the requirements defined for the One-Way Data Link specified in Paragraph 3.1.5.2.

3.1.9 Housekeeping Telemetry

Instrument housekeeping telemetry data shall be transmitted to the spacecraft systems via the Command/Telemetry Bus. This is the same bus described and specified in paragraph 3.1.7 c, Serial Digital Commands/Memory Loads and implementation and operation shall be as specified therein. Housekeeping telemetry provides information to the spacecraft telemetry system on the status and operability of the instrument.

Instrument Housekeeping Telemetry data shall be packetized in a format compatible with the requirements for a Version 1 CCSDS packet.

Spacecraft time shall be incorporated into the housekeeping telemetry data packets as the time of the occurrence of the first measurement of a data unit within the applicable telemetry measurement cycle or epoch.

Housekeeping telemetry data content, formatting and packetizing-/segmentation specifics shall be addressed in the Instrument unique interface document.

3.1.10 Survival Temperature Telemetry Data

Each instrument or instrument separately mounted assembly that contain temperature critical components shall incorporate two independent thermistor circuits. The circuits shall functionally replicate each other. The circuits will not be polled concurrently but are utilized, one, by the spacecraft system, and, the other, by the meteorological communications package dependent on spacecraft operational status.

The electrical characteristics of the thermistor circuits shall be in accordance with the Survival Thermistor Circuit defined in paragraph 3.1.5.5.

3.2 Mechanical Interface Description

The mechanical interface between the Instrument and the spacecraft shall be defined and implemented using the International System of Units (SI) and metric hardware.

3.2.1 Instrument Physical Characteristics

Instrument physical characteristics shall be recorded in the instrument unique interface control document. Subjects to be addressed are the following:

a. Size and Shape. Size and shape shall be established and include any volume required for moving and deployable parts as well as access to clear fields of view.

b. Mass Properties. Mass properties shall be established. The record shall account for all mass states and mass dynamics attributable to deployable, consumable, moving or jettisonable materials or assemblies.

1. Mass. Mass shall be determined to an accuracy of the greater of 0.05 kilogram or 1 percent of instrument mass.

2. Center of Mass. Center of mass shall be determined to an accuracy of less than 5 millimeters spherical error.

3. Moments of inertia. Moments of inertia about each major axis are to be determined for all instrument configurations to an accuracy of 1 percent of the total instrument moment of inertia for that axis.

4. Instrument Induced Disturbances. Dynamic forces and torques induced by instrument operation that are reacted at the instrument to spacecraft mount shall be described and recorded and comply with the limits specified herein. In cases where an instrument is composed of more than one independently mounted assembly, these requirements apply to each independently mounted assembly.

Non recurring transients, events - Forces and torques associated with, and the total momentum imparted to the spacecraft by instrument non-recurring events shall be determined for each axis and their values recorded in the instrument unique interface control document.

Continuous Momentum, Recurring Transient Forces and Torques - For each instrument mounted assembly, disturbance torque, force and momentum shall be determined and comply with the limits calculated in accordance with the method of Figure 5.

c. Sensor fields of View. Fields of view (including required guard-bands) shall be established and defined with regard to location and size of origin, axes, and extent.

d. Thermal Cooler fields of view. Fields of view shall be established and defined with regard to location and size of origin, axes and extent.

e. Instrument Coordinate Reference System. An instrument right-handed orthogonal coordinate reference system shall be defined in the following terms:

The system origin shall be physically located on an accessible, identifiable instrument exterior feature.

The alignment of the axes shall be:
Velocity vector (in orbit plane),
Anti-Sun (defined as Normal to orbit plane), and,
Nadir (in orbit plane).

The right-handed order of the axes shall be :
(Velocity vector) X (anti-Sun) = (Nadir).

3.2.2 Instrument Mounting

a. Instrument Provisions. Instruments shall not require assembly or disassembly to effect mounting on or dismounting from the spacecraft.

Instrument mounting is to be accomplished by means of bolts, passing through instrument flanges, lugs or structural compo-

nents, which mate with spacecraft supplied hardware. At least four bolts shall be used to attach each independently mounted instrument assembly to the spacecraft. Instrument mounting bolts shall be ISO-metric fine thread titanium bolts used with steel alloy washers. Instrument designs shall incorporate mounting provisions of sufficient size, number and location to assure survival of the worst combination of ultimate-level conditions, including thermal differential loading, without permanent deformation or damage to either the instrument or the spacecraft.

Mounting compatibility with the spacecraft will be assured by the use of matched precision drill templates provided by the instrument supplier. Attachment holes shall be located with respect to a reference point with tolerances no greater than the following:

Distance:	± 0.1 mm
Pitch circle	
Radial:	± 0.1 mm
Angle:	± 1 arc minute

The surface flatness of the attachment face shall be less than 0.1 mm per 100 mm. The roughness shall be no more than 1.6 micron R.A.

The attaching bolts and clearance holes shall be normal to the attachment surface with a tolerance of 0.5 degrees.

Instrument assemblies intended for mounting internal to the spacecraft shall use size M4 or larger (M4 preferred) bolts for attachment. Minimum distance between attachments shall be 100 mm. Instruments assemblies (having non-critical alignment) requiring slip-mounts to achieve acceptable attachment loads over the temperature range shall have attachment clearance holes that comply with the following:

Units with maximum distance between attachment holes of ≤ 500 mm shall have an attachment clearance hole diameter equal to the bolt diameter $+0.9$ to 1 mm and an attachment lug thickness of ≥ 5 mm.

Units with maximum distance between attachment holes of >500 mm and ≤ 800 mm shall have an attachment clearance hole diameter equal to the bolt diameter $+1.2$ to 1.3 mm and an attachment lug thickness of ≥ 6 mm.

Units with maximum distance between attachment holes of >800 mm shall address slippage provisions in their instrument unique interface control documents.

Instrument assemblies intended for mounting on the European Spacecraft service module shall be designed to use size M4 bolts for attachment. Minimum distance between attachments shall be 25 mm.

Instrument assemblies intended for mounting external to the European Spacecraft payload module shall be designed to use size M5 or M6 (M5 preferred) bolts for attachment. Minimum distance between attachments shall be 100 mm.

The following information shall be established and recorded in the instrument unique interface control document:

The planarity and location of the instrument mounting surface(s);

The size and number and location of mounting provisions-/bolts;

Requirement for and description of locating pins;

Requirement for and description of shims;

Requirement for and a description of any special mounting hardware provisions or accommodations; and,

Positional and rotational tolerances of the mounted instrument with respect to critical spacecraft axes.

b. Spacecraft Provisions. The spacecraft provides a planar mounting surface to which the instrument is bolted. The surface shall be flat to less than 0.1 millimeters in 100 millimeters and have a surface roughness of 1.6 micron R.A. The mounting surface shall be parallel to one of the spacecraft principle reference planes. The temperature coefficient of expansion of the mounting surface shall be $2.0 \times 10^{-6} (^{\circ}\text{C}^{-1})$ for the purpose of determining instrument attachment loads and designing instrument provisions. Spacecraft instrument-attaching hardware (eg. threaded inserts) shall accommodate axial, radial and torsional loads in accordance with Table 1 without yield or failure. Instrument attachment load limits shall comply with the following:

$$\left(\frac{T}{T_m}\right)^2 + \left(\frac{S}{S_m}\right)^2 + \left(\frac{M}{M_m}\right)^2 \leq 1$$

where:

T, S and M are the actual limit values of Tension, Shear and Moment loads applied to the attachment bolt,

T_m , S_m and M_m are the insert strength capability specified in Table 1.

Note: Compression (C, C_m), where capability is specified, shall be used instead of tension (T, T_m) if a lower instrument load limit would result.

Table 1. S/C Insert Strength for Instrument Mounts				
Location / Application	Tension	Compression	Shear	Moment
	Tm (N)	Cm (N)	Sm (N)	Mm (N-m)
Internal Payload Equipment bay	2350		1615	TBD
Service Module	1000		650	TBD
External payload module				
Nadir pointing panels	2350	1900	2239	TBD
Anti-Nadir pointing panels	900	TBD	1500	TBD
Velocity/anti-velocity panels	2350	1900	1615	TBD
balcony panel	2350	1900	1615	TBD

The instrument mounting design shall be described in the instrument unique interface control document as described in paragraph a. above.

3.2.3 Alignment Provisions

Establishment of instrument alignment shall be accomplished by dimensional control of the instrument mounting hole locations with respect to the critical axes of the instrument. Likewise the locations of mating spacecraft mounting hardware shall be dimensionally controlled with respect to the spacecraft reference frame. Location and tolerances of mounting provisions and critical alignments to be achieved between spacecraft and instrument critical axes shall be established and documented in the instrument unique interface control document.

Instruments requiring precision alignment or alignment knowledge shall be equipped with an optically reflective cube. Optical alignment cubes shall have a surface area on each reference face of at least 645 square millimeters. Reference faces shall be orthogonal to each other within ± 1 arc second.

Optical reference cubes, if used, shall be permanently affixed to the instrument. Alignment cubes shall be located on the instrument such that two faces are accessible for direct viewing by a theodolite mounted external to the spacecraft when the instrument is installed in the spacecraft and without requiring the removal of any other spacecraft mounted instrument, subsystem or hardware item. Optical alignment cube installation location and orientation and direct viewing access paths shall be documented in the instrument unique interface control document.

Instruments requiring alignment cubes must also provide alignment cubes on their drill templates.

3.2.4 Mechanisms

Instruments employing mechanisms shall be designed to minimize collateral effects on the spacecraft. Any collateral effects shall be documented in the instrument unique interface control

document. Mechanisms shall conform to the following interface requirements.

a. Operation. Operation of mechanisms shall produce minimal effects on the static and/or dynamic performance and characteristics of the spacecraft.

b. Caging. Mechanisms requiring caging shall not require power to maintain the caged condition.

c. External Properties. Mechanisms, whose motions are exposed to the external environment, shall not cause unspecified changes to the thermal properties of the spacecraft as a result of repositioning the mechanism.

3.2.5 Harness Tie Points

Instrument provided harness tie points shall be identified, described and located in the instrument unique interface control document.

3.2.6 Connectors

Connectors, connector assignments and keying provisions shall be identified, described and located in the instrument unique interface control drawing. Connectors shall be located such that they can be engaged or disengaged effectively by hand. Connectors shall be accessible for replacement and repair. Removable covers shall be provided for all flight connectors. Connectors not used in flight shall be equipped with captive covers.

3.2.7 Instrument Case Design

Fillets and radii adequate to facilitate cleaning of the exterior of the instrument by solvent wipe methods shall be provided. Mounting surfaces shall be compatible with the mating spacecraft surfaces and shall not oxidize or otherwise deteriorate with time.

3.2.8 Instrument Load Design

All instrument loads shall be reacted at the instrument mounts. Static and dynamic loads may be applied to the instrument in any direction. The instrument shall be capable of operating in a one gravity load applied in any orientation and a zero gravity load environment.

3.2.9 Dynamic Characteristics

Instrument dynamic responses shall have a minimum natural frequency in excess of 100 Hertz. Compliance with this requirement shall be demonstrated by test in accordance with paragraph 10.1.1 of Appendix I.

3.2.10 Instrument Finish

Exposed surfaces shall be compatible with the requirements of the spacecraft thermal control subsystem. Mounting surfaces shall effect electrical bonding with the spacecraft structure, unless other bonding methods are used and documented in the instrument unique interface control document.

3.2.11 Protective Covers

Non-flight removable protective covers required by an instrument during spacecraft integration and handling operations shall be described and documented in the instrument unique interface control document. All protective covers shall be bright red in color and be clearly marked with the words "Remove Before Flight".

3.2.12 Tooling, Fixtures

Instrument required tooling (Eg.: drill templates, special optical devices, etc.) or specialized handling fixtures and devices shall be described and documented in the instrument unique interface control document.

3.3 Thermal Interface Description

The general characteristics of the thermal interface are described below. The instrument thermal control design will evolve from analysis of the unique interactions among the instrument, each spacecraft and the flux environment.

3.3.1 Analytic Tools

a. Instrument Models. Instrument thermal analytic models shall be developed and delivered to the GSFC by the instrument supplier. Models addressing the instrument installation in USA satellites shall be compatible with SINDA and TRASYS software. Models addressing the instrument installation in European satellites shall be compatible with ESATAN and MATRAD software. The computer model documentation shall describe and enumerate for each node the following:

1. Node locations and dimensions with sketches signifying specific regions of the instrument;
2. Surface properties, including solar absorptivity (for externally exposed surfaces) and hemispherical emissivity; where applicable, specular reflectance and transparency;
3. Mass, specific heat and density;
4. Conduction, convection and radiation coefficients and coupling factors; and,
5. Thermal dissipations, with notes describing sources and assumptions.

b. Spacecraft Model. A simplified spacecraft surface model showing all spacecraft surfaces affecting the instrument local environment shall be developed and shall be made available to the instrument developer for analysis.

c. Flux Model. Environmental flux loads on the instrument surfaces for various sun angles shall be determined and shall be made available to the instrument developer for analysis.

3.3.2 Installation

a. Externally Mounted instruments. Instruments that are not enclosed in the spacecraft shall be radiatively and conductively isolated from the spacecraft. The magnitude of the thermal interchange between the instrument and the spacecraft shall not exceed five watts for any normal operating mode or flight attitude. The temperatures of the spacecraft's instrument mounting surfaces shall be within the ranges indicated below.

<u>Instrument Operating Mode</u>	<u>Temperature (degrees Celsius)</u>
Normal instrument Operation	-5 to +30
Normal instrument OFF	-10 to +35
Abnormal Instrument OFF	-20 to +40

Instrument thermal radiator characteristics such as balance temperature, area, location, emissivity, fields of view, shielding requirements and any other interface factors shall be described in the instrument unique interface control document. Instruments, with thermal radiators that are not normally exposed to direct sunlight for proper operation, shall not be damaged by exposure of their radiators to direct sunlight as indicated in paragraph 3.3.3 b, or shall be protected (eg. by the use of re-closeable doors or other means) whenever the instrument is in the safe condition.

b. Internally mounted equipments. Equipments that are enclosed in the spacecraft shall conductively couple to the spacecraft equipment mounting surface. The temperature of the mounting surface shall be within the ranges indicated below.

<u>Instrument Operating Mode</u>	<u>Temperature (Degrees Celsius)</u>
Normal instrument Operation	-5 to +25
Normal instrument OFF	-10 to +35
Abnormal instrument OFF	-20 to +40

Instrument surface flatness and finish (use of thermally conductive fillers is allowed) shall achieve a thermal conductivity of 0.016 watt per square centimeter per degree celsius temperature difference. Surfaces, other than that used for mounting, of internally mounted instruments shall have a thermal emissivity of 0.7 or greater.

3.3.3 Space Thermal Environment

The instruments shall meet their performance requirements while exposed to the operational orbital environment and be capable of meeting all performance requirements after exposure (in the safe condition with survival heater applied) to the non-operating orbital environment described below.

a. Operational Orbital Characteristics. The spacecraft operational orbits are 800 and 824 kilometer, sun-synchronous, circular orbits. The spacecraft attitude is controlled to consistently maintain the Earth pointing axis aligned with nadir, the velocity pointing axis aligned with the velocity vector and the anti-sun axis normal to the shady side of the orbit plane. The orbital sun angle (defined as the angle included between a vector pointing at the sun and the negative anti-sun axis) will range between 0 and 80 degrees during operations. The orbital radiative flux parameters are given below:

<u>Parameter</u>	<u>Hot Case (W/m²)</u>	<u>Cold Case (W/m²)</u>
Solar Constant	1417.5	1285.2
Earth IR Radiance/	264.6/0.25	239.4/0.25
Albedo	248.9/0.3	223.6/0.30
	230/0.35	207.9/0.35

b. Non-Operational Orbital Characteristics. The same as a. above except that the orbital sun angle and spacecraft attitude will not be limited during launch phase through establishment of orbital operational attitude. During post-launch flight phases the spacecraft thermal view-factors may be off-nominal due to the stowed condition of deployables. After the achievement of orbital operational attitude, sun angles may exceed operational limits during maneuvers or anomalous conditions. Instrument radiators, not normally exposed to direct sunlight, may, for a period not exceeding 14 minutes duration within each 24 hour period, be exposed to direct sunlight during maneuvers or anomalous conditions.

3.3.4 Electrical Heaters.

Use of heaters shall be described and documented in the instrument unique interface control document.

a. Operational Heaters. The Instrument may use heaters to achieve active thermal control. Operational heaters shall utilize the main power bus and shall operate within the power budget allocated for the instrument.

b. Survival Heaters. Survival heaters may be incorporated into the instrument. These heaters shall be activated by the instrument (e.g. by use of thermostats) to maintain the non-operating instrument at a temperature above its damage threshold and to maintain the command/response terminal of the instrument at its minimum operating temperature. Survival heaters shall be sized to meet performance requirements in both operational and

non-operational orbital conditions and in the OFF state. Survival heaters shall be powered by the survival heater bus which is active when the main bus power is not applied.

3.4 Magnetic Interface

The instruments shall meet all performance requirements when exposed to the magnetic environment defined below. Conformance with this requirement shall be demonstrated by the methods specified in paragraph 10.1.9.2 of Appendix I.

3.4.1 Spacecraft Magnetic Fields

The spacecraft generates magnetic fields during orbital operations. The maximum spacecraft magnetic moment with which the instruments must interact is 25 Ampere-Meters² per axis. The maximum static magnetic field with which the instruments must interact is 155 dB_{PT} which may occur at any intercept angle.

3.4.2 Instrument Characteristics

Instruments shall by design minimize susceptibility to and the emission of stray magnetic fields. Instruments, and instrument components where appropriate, shall meet operating requirements when exposed to the magnetic environment of the spacecraft. Instruments for which magnetic containment or immunity requires special spacecraft accommodation shall have described and characterized the requirements for such in the instrument unique interface control document.

Instruments not taking an exception to this practice in their instrument unique interface control document will be degaussed prior to installation on the spacecraft. Degaussing field strength will be 5×10^{-3} tesla maximum at a frequency of 0.5 Hertz.

3.4.3 Instrument Magnetic Properties

Instrument magnetic properties shall conform to the following requirements. Note: Dipole moments, when specified, are as though determined from magnetic field measurements conducted at a distance of three times the maximum linear dimension of the item under test. Conformance with these requirements shall be demonstrated by the methods specified in paragraph 10.1.9.1 of Appendix I.

a. Initial Perm Test. The maximum D.C. magnetic moment of the instrument following manufacture shall not exceed 0.2 ampere-meters².

b. Perm Levels After Exposure to Magnetic Field. The maximum magnetic moment of the instrument after exposure to magnetic field test levels of 15×10^{-4} tesla shall not exceed 0.3 ampere-meters².

c. Perm Levels After Exposure to Deperm Test. The maximum magnetic moment of the instrument after exposure to magnetic field deperm levels of 50×10^{-4} tesla shall not exceed 0.1 ampere-meters²

d. Induced Magnetic Field Characteristic Measurement. The induced magnetic field of the instrument shall be measured while the instrument is turned off and exposed to a magnetic field test level of 0.6×10^{-4} tesla. The measurement shall be made by a test magnetometer that can null the test magnetic field. The results of this test shall be recorded in the instrument test data record.

e. Stray Magnetic Field Measurement. The instrument magnetic moment shall not change in excess of 0.05 ampere-meters² due to internal current flows.

3.5 Electromagnetic Compatibility Requirements

The general requirements for electromagnetic compatibility are as follows:

a. The spacecraft and its elements shall not generate electromagnetic interference that could adversely affect its own subsystems or components, other payloads, or the safety and operation of the launch vehicle or launch site.

b. The spacecraft and its subsystems and components shall not be susceptible to emissions that could adversely affect their safety and performance. This applies whether the emissions are self generated or emanate from other sources, or whether they are intentional or unintentional.

3.5.1 Requirements Summary

The EMC requirements defined herein are intended to envelope the environment that may be expected during a typical mission and allow for some degradation of the hardware during the mission. The requirements also envelope the environments usually encountered during integration and ground testing. The electromagnetic environment is implicit in the constraints placed on electromagnetic emission characteristics and susceptibilities of the instruments and/or their major components.

3.5.1.1 Range of Requirements

Table 2 lists the EMC requirements and related tests that apply to the instrument. The EMC test program is intended to uncover workmanship defects and unit-to unit variations in electromagnetic characteristics, as well as design flaws, therefore, all flight hardware shall be tested.

The EMC tests are intended to verify that:

(1) The hardware will operate properly if subjected to conducted or radiated emissions from other sources that could occur during launch or in orbit.

(2) The hardware does not generate either conducted or radiated energy that could hinder the operation of other systems.

Table 2. Applicable EMC Requirements

<u>Type(1)</u>	<u>Requirement/Test</u>	<u>Paragraph</u>	<u>Figure</u>	<u>Mil-Std-461 (Ref)</u>
CE	Regulated Power Leads	3.5.2.1.1	N/A	CE01, CE03
CE	Unregulated Power Leads	3.5.2.1.2	6	CE01, CE03
		3.5.2.1.2	7	CE03
CE	Signal Leads and Bundles	3.5.2.1.3	8	CE03
CE	Antenna terminals	3.5.2.1.4	9	CE06
RE	AC Magnetic Fields	3.5.2.2 a	N/A	RE04
RE	E-Fields	3.5.2.2 b	10	RE02
		3.5.2.2 c	Table 3	RE02
		3.5.2.2 d	11	RE02
RE	Spurious (Transmit Antenna) (2)	3.5.2.2 e	9	RE03
CS	Regulated Power Leads	3.5.3.1.1 a	N/A	CS01, CS02
CS	Reg Pwr Lead, Transients	3.5.3.1.1 b	N/A	CS06
CS	Unregulated Power Leads	3.5.3.1.2 a	12	CS01, CS02
CS	Unreg Pwr Lead, Transients	3.5.3.1.2 b	N/A	CS06
CS	Intermod Products (3)	3.5.3.1.3	N/A	CS03
CS	Signal rejection (3)	3.5.3.1.4	N/A	CS04
CS	Cross Modulation (3)	3.5.3.1.5	N/A	CS05
RS	E-Field	3.5.3.2 a	13	RS03
RS	Magnetic Fields (4)	3.5.3.2 b	14	N/A

Notes:

- 1. CE - Conducted Emissions
- RE - Radiated Emissions
- CS - Conducted Susceptibility
- RS - Radiated Susceptibility
- N/A - Not Applicable

2. Alternative test method to para. 3.5.2.1.4.

3. Applies to receivers and tuned components only.

4. Applies to magnetically sensitive instruments, subsystems and components only.

3.5.1.2 Basis of the Tests

A description of the individual EMC requirements listed in Table 2 including their limits and related test for conformance procedures, are provided in paragraphs 3.5.2 through 3.5.3.2. Most of the tests are based on the requirements of MIL-STD 461C and MIL-STD-462, as amended by Notice 1, and MIL-STD-463A. The MIL-STD limits have in some cases been modified to meet the EMC requirements of the common instrument applications.

3.5.1.3 Corrective Action

Emitted spurious signals that are stronger than the testing limits shall be eliminated. Spurious signals that are weaker than specified limits shall be analyzed to determine if a subsequent change in frequency or amplitude can occur. If this is the case, the spurious signal shall be eliminated to protect the payload and instrument from the potential for interference. Susceptibilities to the specified induced electromagnetic environments shall be corrected. Retest shall be performed to verify that the implemented solutions are effective.

3.5.2 Emission Requirements

The instrument electromagnetic emissions shall not exceed the limits defined herein.

3.5.2.1 Conducted Emission Limits

3.5.2.1.1 Regulated Power Circuits - Instrument conducted emissions on regulated power circuits shall comply with the limits defined in paragraph 3.1.1.3.1 for the main power circuit and paragraph 3.1.1.3.2 for the pulse load power circuit.

Conducted emission limits defined for the main power circuit include peak to peak steady state ripple current, instantaneous rate of change in currents, and transient current limits (including turn-on).

Conducted emission limits defined for the pulse load power circuit include peak to peak steady state ripple current, instantaneous rate of change in currents, and pulse peak current and integrated charge.

Test methods of Mil-Std-462, CE01 and CE03, shall be utilized. An oscilloscope with a direct current measuring capability that has a minimum bandwidth of 30 MHz may be substituted for the EMI meter specified when making compliance determining measurements.

3.5.2.1.2 Unregulated Power Circuits - Conducted emission limits on unregulated instrument power circuits are defined below.

- a. Narrowband conducted emissions CE01, CE03 (Powerlines) - Narrow band conducted emissions on power and on power return lines shall be limited to the levels specified in Figure 6. Testing for conformance shall be in accordance with Mil-Std-

461C and 462, test numbers CE01 and CE03, as applicable, with the limits as shown in Figure 6.

b. Broadband conducted emissions CE03 (Powerlines) - Broadband conducted emissions on power and on power return lines shall be limited to the levels specified in Figure 7. Testing for conformance shall be in accordance with Mil-Std-461C and 462, test number CE03, with limits as shown in Figure 7.

3.5.2.1.3 Signal Leads and Bundles - Unintentional narrow band conducted emissions in the frequency range of 20 Hz to 50 MHz shall not appear on control and signal leads (common mode) in excess of the values shown in Figure 8. Limits are specified for individual signal leads (or signal pairs) and cable bundles and are to be applied as appropriate to the unit under test. Testing for conformance shall be in accordance with Mil-Std-461C and 462, test number CE03, as applicable, with the limits as shown in Figure 8.

3.5.2.1.4 Conducted emissions CE06 (Antenna Terminals) - Conducted emissions on the antenna terminals of instrument receivers and transmitters in key-up modes shall not exceed 34dB μ Volts for narrowband emissions and 40 dB μ Volt/MHz for broadband emissions. Harmonic and all other spurious emissions from transmitters in the key-down mode shall not exceed the limits shown in Figure 9. Testing shall be in accordance with Mil-Std-462, test number CE06. The test shall be conducted on all receivers and transmitters before they are integrated with their antenna systems.

3.5.2.2 Radiated Emission Limits

Radiated emission limits applied to the instrument are defined below.

a. Radiated AC Magnetic Fields RE04. Radiated AC magnetic field levels produced by the instrument shall be limited to 60 dBpT from 20 hertz to 50 kilohertz. Testing for conformance shall be in accordance with Mil-Std-462, test number RE04, with the limits as defined above.

b. Radiated Narrow Band Electric Fields RE02. Unintentional radiated narrowband electric field levels produced by the instrument shall not exceed the levels specified in Figure 10. Testing for conformance shall be in accordance with Mil-Std-461C and 462, test number RE02, with the test frequency range and limits as defined in Figure 10.

c. Additional Requirements RE02. In addition to the requirements stated above, narrowband radiated emissions produced by the instrument shall not exceed the limits defined in Table 3 for the specified frequency bands. Measurements for compliance with these requirements shall be made in accordance with method RE02 with the EMI meter replaced by a MITEQ preamp (AU-2A-0550)

and a spectrum analyzer (HP 8566A) or equipments of equivalent capability. The instrument under test and the associated clock and control signals shall have power applied. The difference in analyzer levels shall be noted for both white noise and spurious signals.

The test antenna shall be tuned to the center of each of the four bands defined in Table 3. Prior to making actual measurements, the test antenna shall be demated and the cable terminated with fifty ohms. The noise floor of the equipment shall be verified to be lower than the specified signal limit for each of the measurements. The resolution bandwidths which correlate with the specified limits for the measurements below the frequency of 1 GHz are: 100 Hz for both the -150 dB_m and the -145 dB_m measurements, 1 kHz for the -125 dB_m measurement, and 3 kHz for the -100 dB_m measurement. For the frequency band above 1GHz the resolution bandwidth is 2 kHz.

Table 3. Additional Narrowband Radiated Emission Limits

MAX SIGNAL LEVEL (dBm)	121.5 MHz BAND	243 MHz BAND	406.05 MHz BAND	401.650 MHz BAND
-100	118.000-120.000	236.000-240.000	385.100-401.100	
-125	120.000-121.450	240.000-242.925	401.100-405.900	396.000-401.500
-145	121.450-121.485	242.925-242.975	405.900-406.000	401.500-401.600
-150	121.485-121.515	242.975-243.025	406.000-406.100	401.600-401.700
-145	121.515-121.550	243.025-243.075	406.100-406.200	401.700-401.800
-125	121.550-123.000	243.075-246.000	406.200-411.000	401.800-406.000
-100	123.000-125.000	246.000-250.000	411.000-425.000	

Radiated emissions in the frequency range from 2010 Megahertz to 2040 Megahertz shall be less than -120 dBm.

d. Radiated Broadband Electric Fields RE02. Unintentional radiated broadband electric field levels produced by the instrument shall not exceed the levels specified in Figure 11. Testing for conformance shall be in accordance with Mil-Std-461C and 462, test number RE02, with the test frequency range and limits as defined in Figure 11.

e. Radiated spurious and harmonic emissions RE03 (transmitter antennas). Radiated spurious and harmonic emissions from instrument transmitter antennas shall not exceed the limits of Figure 9, the same limits as those for conducted emissions on antenna terminals defined in paragraph 3.5.2.1.4. When Mil-Std-462 test CE06 for conducted emissions on antenna terminals cannot be applied, test RE03 for radiated spurious and harmonic emissions shall be used as an alternative test.

3.5.3 Susceptibility Requirements

The instrument shall meet its design requirements when subjected to the electromagnetic susceptibility test environments and levels defined herein.

3.5.3.1 Conducted Susceptibility Requirements

Instruments shall not be susceptible to conducted interference introduced on the power and power return circuits at the levels indicated herein.

3.5.3.1.1 Regulated Power Circuits - The instrument shall meet its performance requirements when exposed to the following levels of interference superimposed on the main power circuit and the pulse load power circuit and their respective returns.

a. Conducted Susceptibility CS01, CS02 (Powerlines) - Test methods of Mil-Std-462, CE01 and CE03, shall be utilized.

The injected level of the carrier signal superimposed on the main power circuit and its return shall be 300 mv peak to peak.

The injected level of the carrier signal superimposed on the pulse load power circuit and its return shall be 400 mv peak to peak.

The carrier frequency shall be swept through the range of 30 Hz to 150 kilohertz. The superimposed carrier signal shall be modulated as defined below:

In the range of 30 Hertz to 2000 Hertz the interfering carrier signal shall be unmodulated;

In the range of 2000 Hertz to 150 kilohertz the interfering carrier signal shall be amplitude modulated with a 1000 Hertz sine wave with a modulation depth of 50 percent.

b. Conducted Susceptibility CS06 (Powerline Transient) - Test methods of Mil-Std-462, CE06 shall be utilized. A test transient waveform signal shall be repetitively applied to each instrument power circuit.

The transient voltage superimposed on the main power circuit voltage shall have peak amplitudes of +10 and -12 volts.

The transient voltage superimposed on the pulse load power circuit voltage shall have peak amplitudes of +8 and -13 volts.

The transient duration shall be 10 μ s and the waveform characteristics shall be those defined in test method CS06 as Spike #1. The transients shall be applied for a duration of ten minutes at a repetition rate of ten spikes per second for each polarity of spike .

3.5.3.1.2 Unregulated Power Circuits

a. Conducted Susceptibility CS01, CS02 (Powerlines) - The tests shall be conducted over the frequency range of 30 Hertz to 400 Megahertz in accordance with the limit requirements specified in Figure 12 and the test procedures of Mil-Std 461C and 462, tests CS01 and CS02. If degraded performance is observed, the signal level shall be decreased to determine the threshold of interference. Modulation shall be applied to the Susceptibility signal as defined below:

In the range of 30 Hertz to 2000 Hertz the signal shall be unmodulated;

In the range of 2000 Hertz to 400 Megahertz the signal shall be amplitude modulated with a 1000 Hertz sine wave with a modulation depth of 50 percent.

b. Conducted Susceptibility CS06 (Powerline Transient) - A test transient waveform signal shall be repetitively applied to each instrument powerline in accordance with Mil-Std-461C and 462, test method CS06, as modified herein. The applied transient shall equal the powerline D.C. voltage, the resulting peak voltage being twice the powerline D.C. level. The transient duration and waveform characteristics are those defined in test method CS06 as Spike #1. The transient shall be applied for a duration of five minutes at a repetition rate of sixty pulses per second.

3.5.3.1.3 Two-Signal Intermodulation CS03 - This test requirement applies to receivers and tuned amplifiers, at the component level of assembly, that operate in the frequency range of 30 Hz to 18 GHz. The items shall not exhibit any intermodulation responses when tested in accordance with Mil-Std-461C and Mil-Std-462, test number CS03 as amended herein. CS03 is amended to increase the test frequency range upper limit to 40 GHz.

3.5.3.1.4 Rejection of Undesired Signals CS04 - This test requirement applies to receivers and tuned amplifiers, at the component level of assembly, that operate in the frequency range of 30 Hz to 18 GHz. The items shall not exhibit any spurious responses when tested in accordance with Mil-Std-461C and Mil-Std-462, test number CS04 as amended herein. CS04 is amended to increase the test frequency range upper limit to 40 GHz.

3.5.3.1.5 Cross Modulation CS05 - This test requirement applies to receivers and tuned amplifiers that operate in the frequency range of 30 Hz to 18 GHz. The items shall not exhibit any cross modulation responses when tested in accordance with Mil-Std-461C and Mil-Std-462, test number CS05 as amended herein. CS05 is amended to increase the test frequency range upper limit to 40 GHz.

3.5.3.2 Radiated Susceptibility Requirements

The following radiated susceptibility tests shall be applied to the instrument. The tests are based on M1-Std-461C and 462, as supplemented.

a. Radiated Susceptibility Test RS03 (E-Field). The instrument shall be exposed to external electromagnetic signals in accordance with the requirements and test methods of RS03. The instrument shall meet its performance requirements while exposed to the E-Field EMI levels specified in Figure 13 and in Table 4. below.

Table 4. Discrete Frequency E-Field EMI Levels

<u>Frequency (MHz)</u>	<u>E-Field (V/m)</u>
137.1 ₁	40
137.9 ₁	40
460.0	10
468.0	40
1544.3	40
1701.0 ₂	40
1707.0 ₂	40
2247.5 ₂	40
5300.0	162
7500.0	40

1. Frequencies are not applied simultaneously.
 2. Frequencies are not applied simultaneously.
-

b. Magnetic Field Susceptibility. The instrument shall meet its performance requirements while exposed to the A.C. magnetic field EMI levels specified in Figure 14. Conformance with this requirements shall be demonstrated by the methods specified in paragraph 10.1.9.2 of Appendix I.

3.5.3.3 Direct Arc Discharge Susceptibility Requirements

Instruments shall exhibit no malfunction, degradation of performance or deviation from specified parameters beyond normal tolerance limits when the instrument and/or attached harness cables are exposed to direct repetitive arcing discharge of up to 10 millijoules per arc of energy. Test for conformance with these requirements shall be demonstrated by the methods specified in paragraph 10.1.12 of appendix I.

3.6 Structural/Mechanical Environmental Interface.

The Instrument shall meet all structural and operational performance requirements during (or after, if applicable) exposure to anticipated natural, self induced or externally induced environ-

ments. These may be induced in the course of prelaunch transportation and handling, launch, and orbital operations. Likewise the instrument shall not itself be a source of dynamic disturbances that could be deleterious to the survival or operation of the spacecraft, or any other equipment.

The compatibility with the dynamic environment shall be assured by a program of analytic and test activities addressing the structural and mechanical durability and behavior of the instrument. The Instrument development program shall adhere to the Structural and mechanical verification requirements defined in Appendix I with the limits specified herein.

a. Flight environment - The instrument shall meet all structural and functional requirements during or after exposure to the induced environments as specified herein.

b. Transportation, handling and storage - These environments are controllable and shall be limited to values less severe than those experienced during flight environments of para a. above. Institutional requirements for achieving these controlled environments shall be determined and implemented by the instrument developer.

3.6.1 Non-Operating Environments.

The Instrument shall operate within specified limits after exposure to any natural combination of the non-operating environments specified below.

a. Pressure/Pressure Profile. Ambient pressure limits are 1.05×10^5 Pascal to less than 1×10^3 Pascal. The launch pressure release profile is specified in paragraph 10.1.6 of Appendix I.

b. Temperature. Terrestrial and orbital non-operating temperature limits shall be determined by the method described in paragraph 3.3. Instrument performance shall be validated by the thermal balance test specified in paragraph 10.1.10 of appendix I.

c. Relative Humidity. 0 to 80 percent.

d. Fungus. The equipment shall not support fungus growth.

e. Loads. Non Operating load limits are specified in para. 10.1.2 and Figures 10.1 and 10.2 of Appendix I.

f. Shock. Non-Operating shock limits are specified in paragraph 10.1.4 and Figure 10.4 of Appendix I.

g. Vibration. Non-Operating random and sinusoidal vibration limits are specified in paragraph 10.1.3 and Figure 10.3 of Appendix I.

h. Acoustic. The non-operating acoustic environment is specified in paragraph 10.1.5 and Table 10.2 of appendix I

3.6.2 Operating Environments.

The Instrument shall operate within specified limits during exposure to any natural combination of operating environments specified herein.

a. Pressure. 67×10^3 to 1.05×10^5 pascal, and less than 1×10^{-3} pascal.

b. Temperature. Terrestrial and orbital temperature limits shall be determined by the method described in paragraph 3.3 and instrument performance and durability validated by the thermal balance test specified in paragraph 10.1.10 and the thermal-vacuum test series specified in paragraph 10.1.11 of Appendix I.

c. Relative Humidity. 0 to 80 percent

d. Fungus. The equipment shall not support fungus growth.

e. Acceleration. 0 to 1 gravity load, any orientation.

f. Vibration. Negligible.

g. Acoustic. Negligible.

h. Shock. Negligible.

i. Ionizing Radiation. Environmental limits shall be determined in accordance with paragraph 3.8. Instrument performance shall be validated by the test requirement of paragraph 10.1.8 of Appendix I.

3.7 Contamination control

a. Instruments. Instruments shall be contamination free to the extent that they do not degrade the performance or cleanliness of the spacecraft or any other equipment. Cleanliness shall be achieved by the judicious selection of non-contaminating components and materials, use of facilities and processes that will minimize contamination from these sources and use of processes to clean equipments of any accrued contaminants. A contamination control program addressing:

1. Determination of contamination sensitivity,
2. Determination of contamination allowances,
3. Determination of a contamination budget, and,
4. Development and implementation of a contamination control procedures;

shall be effected.

Instrument materials and design approaches shall be selected in accordance with criteria defined in their respective development specifications. In addition the following criteria addressing the contamination aspects of the interface shall be observed:

Polymer Materials. Surfaces sensitive to contamination shall not be exposed to the outgassing products of polymeric materials (particularly if such materials have temperatures above 40 degrees C.). Materials used shall meet the requirements of NASA RP 1124, Outgassing Data for Selecting Spacecraft Materials, for total mass loss not exceeding 1% and collected volatile condensible material not exceeding 0.1%. Protective measures shall be provided for critical surfaces that are sensitive to the foregoing requirement.

Lubricants. Lubricants shall not be exposed to outgassing products or other materials that are incompatible with the lubricant. Lubricant shall be effectively contained by design practice (Eg.: use of labyrinths and barrier films). Lubricants should be transparent at mission wavelengths. Lubricant quantity shall be sufficient to exceed mission life requirements (including storage) by a safety margin of 0.5. The adequacy of the lubricant design shall be demonstrated by test and analyses.

Dissimilar Metals. The use of dissimilar metals in direct contact shall be avoided where ever possible. Where dissimilar metals are used, protection against electrolytic corrosion shall be applied in accordance with Mil-Std-454, Requirement 16.

Propellant Compatibility. All spacecraft materials exposed to fumes, spillage, and combustion products of propellants shall be selected (or protected) to negate any degradation of physical or mechanical properties. These materials shall be rated with respect to corrosion rate, stress corrosion cracking, effect on fluid decomposition, and effect on autogenous temperature.

Cold Flow. The use of non-metallic materials that are subject to cold flow under load shall be permitted only with designs that effectively restrain the material.

b. Spacecraft. Spacecraft assembly, integration and test operations shall be performed in an environment conforming to Fed-Std-209B, Class 100,000 or better. Lower level assemblies, including instruments shall be cleaned prior to installation in the spacecraft. A contamination control plan shall be in effect wherein contaminating materials will be kept outside the controlled area. Molecular contaminants in the spacecraft assembly area shall be continuously monitored and accretion rates controlled or negated by periodic cleaning.

3.8 Ionizing Radiation Environment

a. Total Dose. Instruments shall meet their specified performance when exposed to the radiation environment defined in

NASA X-600-87-11, Metsat Charged Particle Environment Study for the specified orbit, useful life and worst case time period. These values shall be determined for inclusion in the instrument specific interface control document.

b. Single Event Upset. Instruments shall be capable of withstanding and recovering from single event upsets and transients induced by the singular or combined effects of cosmic rays, solar flares, and geomagnetically trapped protons.

Figures

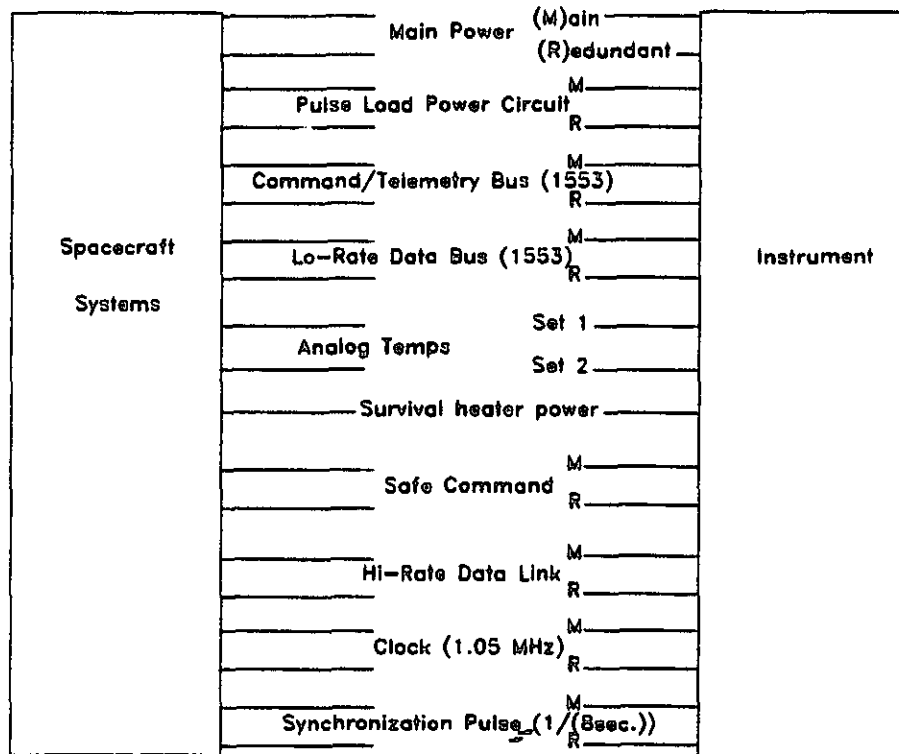


Figure 1. Instrument/Spacecraft Electrical Interface

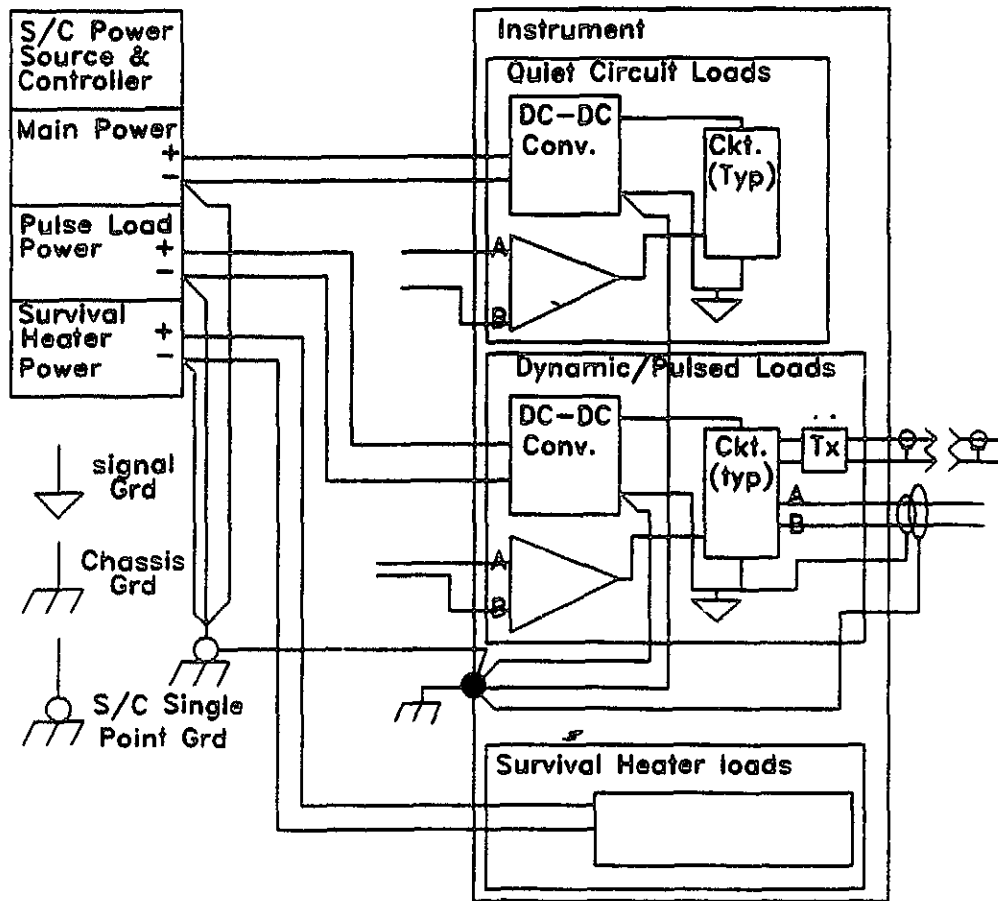
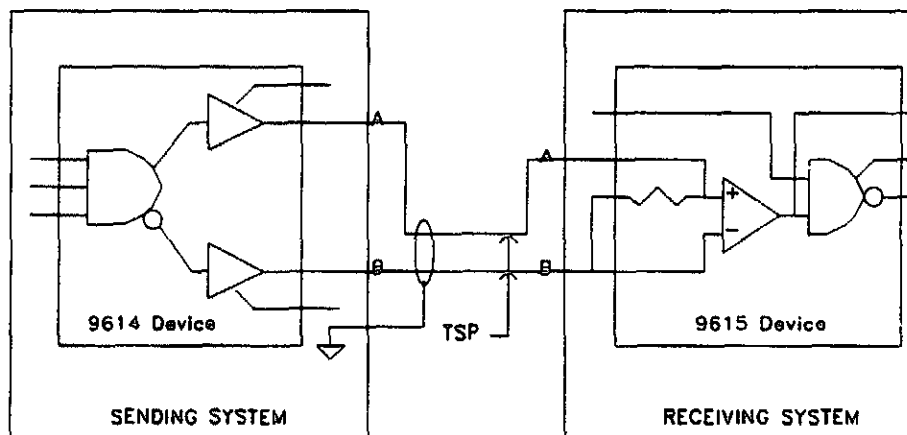
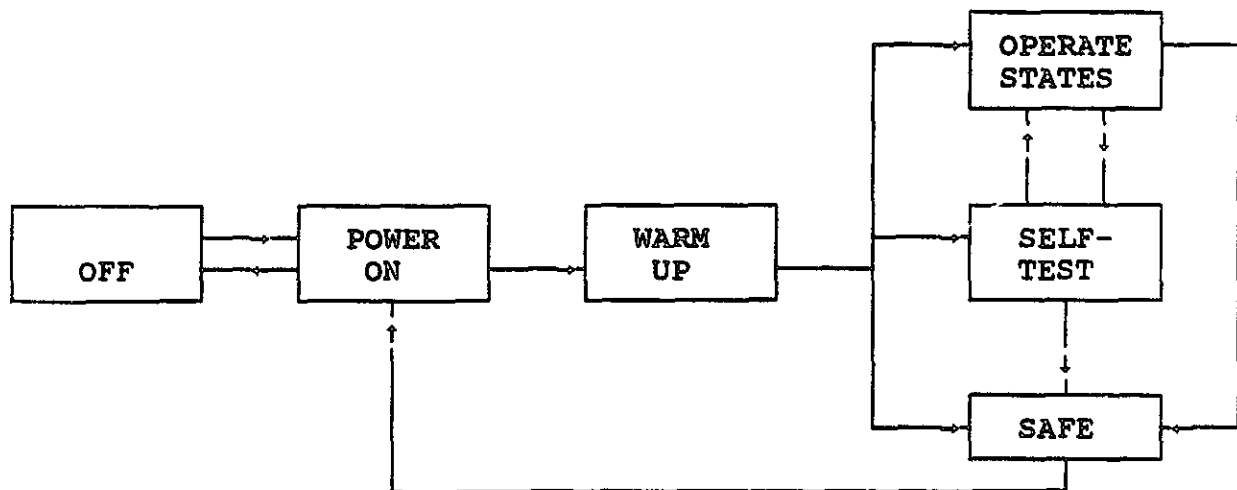


Figure 2. Single Point Ground Depicted



Note: Logic "ON": "A" signal more positive than "B" Signal
Logic "OFF": "B" signal more positive than "A" signal

Figure 3. Bi-Polar Discrete and Clock Signal I/F Depicted



OFF State

Survival Heater Power Only
Equipment mechanically safed
Exits to POWER-ON by application of main power

POWER-ON State

Terminal Active
Operational Heaters On
Equipment mechanically Safed
Status message indicates state and safe
Entered from OFF by application of power
Entered from SAFE at completion of Mechanical safing procedures
Exits to WARM-UP by command
Exits to OFF by removal of power

WARM-UP State

Unit spins up, warms up, conditions sensors, opens doors, etc.
Status message indicates state, not safed, and ready or not ready
Exits to OPERATE or SELF-TEST by command
Exits to SAFE by SAFE discrete or command

OPERATE State(s)

Entered from WARM-UP or SELF-TEST by command
Status message indicates state, not safed
Exits to SELF-TEST by command
Exits to SAFE by SAFE discrete or command

SELF-TEST State

Runs self test, alignment or calibration activities
Status message indicates state, not safed
Entered from WARM-UP or OPERATE by command
Exits to OPERATE by command
Exits to SAFE by SAFE discrete or command

SAFE State

Mechanically and electrically safes the unit preparatory to shut down (e.g. parks mirrors, closes cooler doors, etc.)
State is always entered as prelude to shut down
Status message indicates state and safed or not safed
Entered from any unsafed state by SAFE discrete or command or default (eg. command to return to POWER-ON state)
Exits automatically to POWER-ON state when safing procedures accomplished

Figure 4. Conceptual Instrument State Diagram

For each instrument-mounted assembly, disturbance torque, force and momentum shall be determined and limited in accordance with the following calculations¹:

- For disturbances in the frequency range $0 < \omega < 0.062$ rad/sec (the control bandwidth), let $S_1(\omega)$ be the power spectrum of the disturbance momentum vector $(h_v, h_a, h_n)^T$, and

$$G_1(\omega) = \begin{bmatrix} 0 & 0 & 2.4(10^{-3}) \\ 0 & 3.2(10^{-2})j\omega & 0 \\ 2.4(10^{-3}) & 0 & 0 \end{bmatrix}, \quad j = \sqrt{-1}.$$

- For disturbances in the frequency range $0.062 \leq \omega < 0.8$ rad/sec (rigid body regime outside the control bandwidth), let $S_2(\omega)$ be the power spectrum of the disturbance torque/force vector $(\tau_v, \tau_a, \tau_n, f_v, f_a, f_n)^T$, and

$$G_2(\omega) = \frac{1}{\omega^2} \begin{bmatrix} 1.6(10^{-4}) & 0 & 0 & 0 & 8.7(10^{-5}) & 4.0(10^{-5}) \\ 0 & 1.5(10^{-4}) & 0 & 8.5(10^{-5}) & 0 & 2.0(10^{-4}) \\ 0 & 0 & 8.8(10^{-5}) & 2.2(10^{-5}) & 1.1(10^{-4}) & 0 \end{bmatrix}.$$

- For disturbances in the frequency range $0.8 \leq \omega < \infty$ rad/sec (the regime of flexible-body modes), let $S_3(\omega)$ be the power spectrum of the disturbance torque/force vector $(\tau_v, \tau_a, \tau_n, f_v, f_a, f_n)^T$, and

$$G_3(\omega) = \frac{1}{\omega^2} \begin{bmatrix} 2.2(10^{-2}) & 5.6(10^{-5}) & 6.7(10^{-4}) & 1.7(10^{-4}) & 2.0(10^{-3}) & 8.5(10^{-3}) \\ 6.7(10^{-5}) & 3.1(10^{-4}) & 7.7(10^{-4}) & 4.9(10^{-4}) & 5.7(10^{-6}) & 1.6(10^{-5}) \\ 4.5(10^{-4}) & 7.7(10^{-4}) & 3.1(10^{-3}) & 2.0(10^{-3}) & 5.7(10^{-5}) & 2.0(10^{-4}) \end{bmatrix}.$$

- Compute the following summation over all discrete frequencies ω_k at which spectral lines exist, including negative frequencies since $S(\omega_k) = S(-\omega_k)$:

$$C_i = \sum_{k=-\infty}^{\infty} G_i(\omega_k) S_i(\omega_k) G_i^T(-\omega_k), \quad i = 1, 2, 3.$$

To ensure compliance with spacecraft pointing and jitter requirements, all of the following conditions shall be met:

1. The total spacecraft pointing response to disturbances shall be within the following limit:

$$\begin{pmatrix} \sigma_v^2 \\ \sigma_a^2 \\ \sigma_n^2 \end{pmatrix} = \text{diag}(C_1 + C_2 + C_3) < \begin{pmatrix} 5.0(10^{-10}) \\ 5.0(10^{-10}) \\ 5.0(10^{-10}) \end{pmatrix} \text{ radians}^2$$

2. Momentum about the pitch axis at any frequency shall be less than 0.5 Nms.
3. Bias momentum about roll and yaw shall be limited to 0.5 Nms.

¹All units are metric - N, Nm, Nms, etc. - and radians. The roll, pitch, and yaw axes are denoted respectively, v (velocity), a (anti-sun), and n (nadir). Elements of G_1 have units of rad/(N m s) and elements of G_2 and G_3 have units rad/(kg·m²) in the left three columns and units rad/(kg·m) in the right three columns. The 3 x 3 matrix S_1 has units (Nms)². The 6 x 6 matrices S_2 and S_3 have units (Nm)² for the first three diagonal elements and units N² for the last three diagonal elements. In general, the power spectrum matrices S_1 , S_2 , and S_3 are usually diagonal, i.e., the cross power spectra are usually assumed to be zero.

Figure 5. Evaluation of Instrument-generated Disturbances

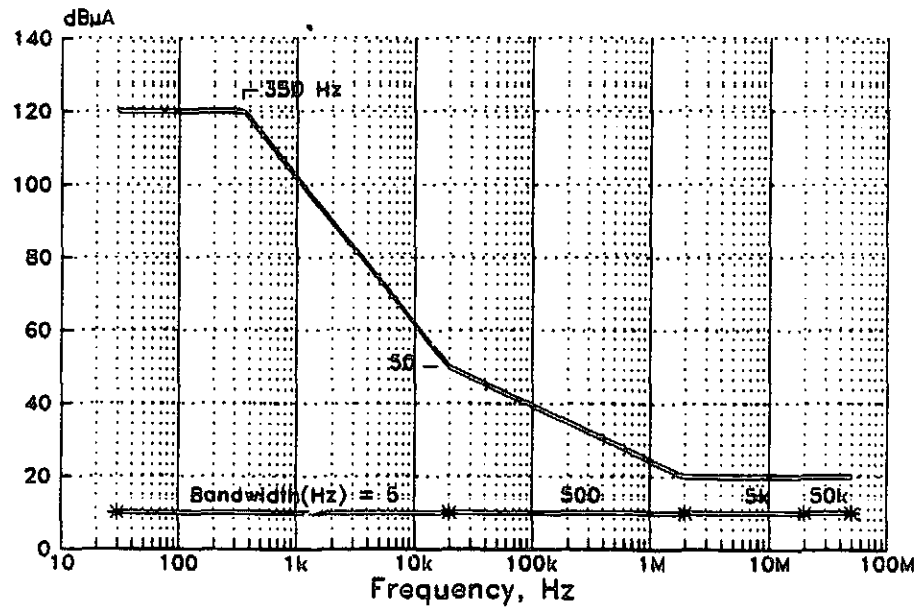


Figure 6. Narrowband Conducted Emission Limits (Powerlines)

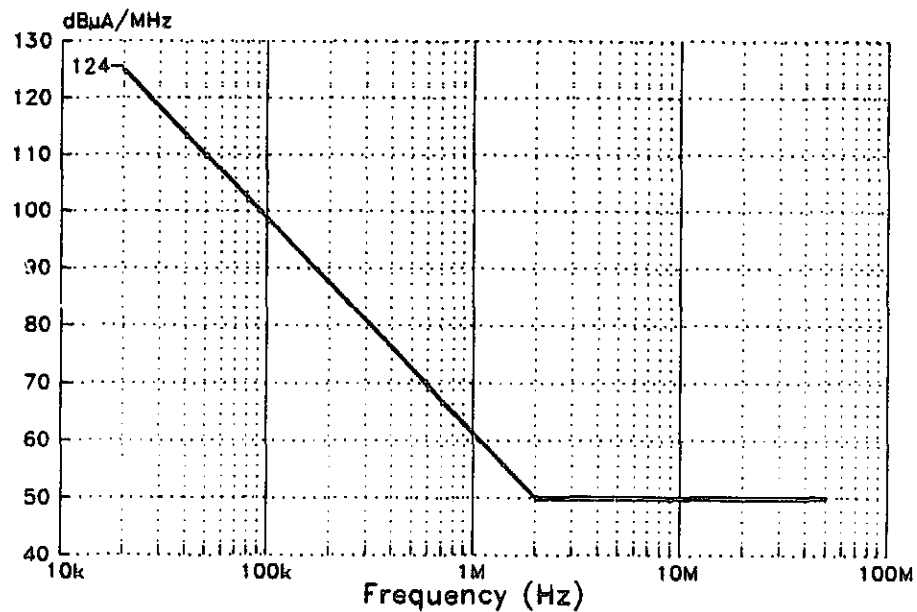


Figure 7. Broadband Conducted Emission Limits (Powerlines)

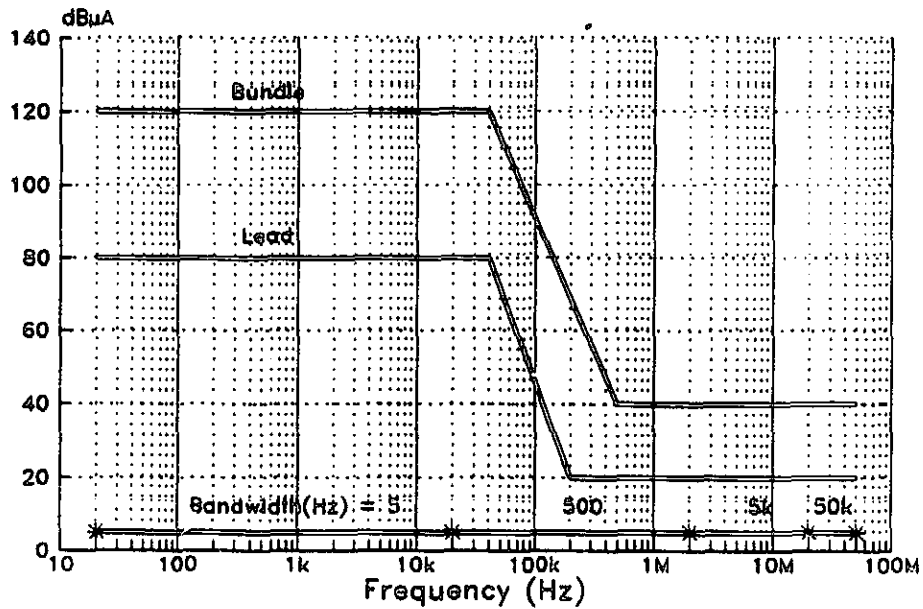


Figure 8. NB Conducted Emission Limits (Signal Leads and Bundles)

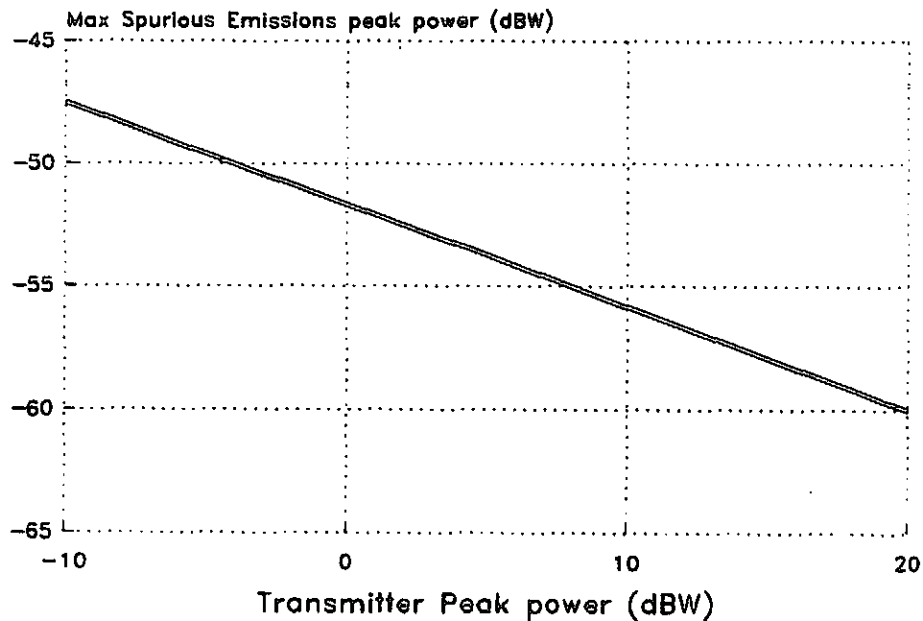


Figure 9. Harmonic and Spurious Emission Limits

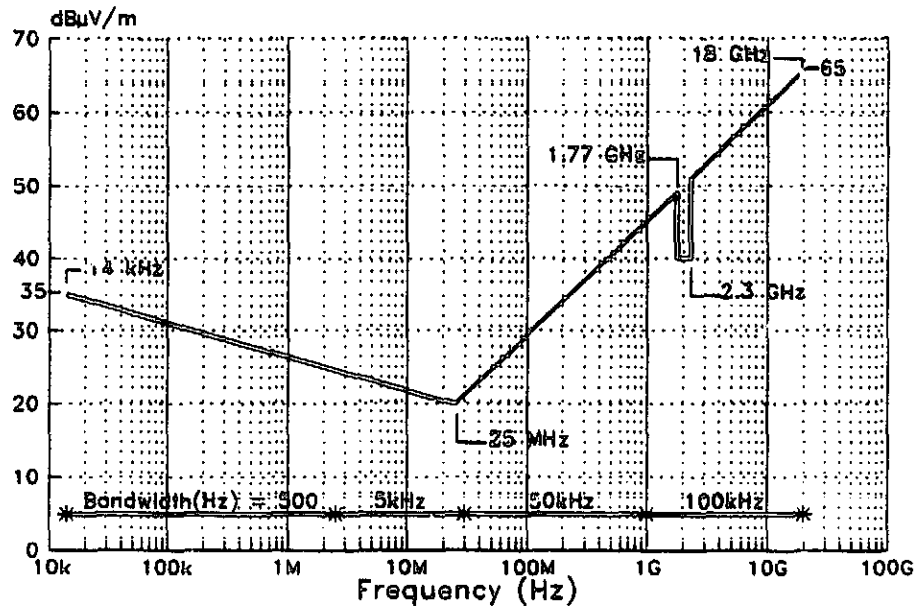


Figure 10. Limits, Unintentional NB Radiated E-Field Emissions

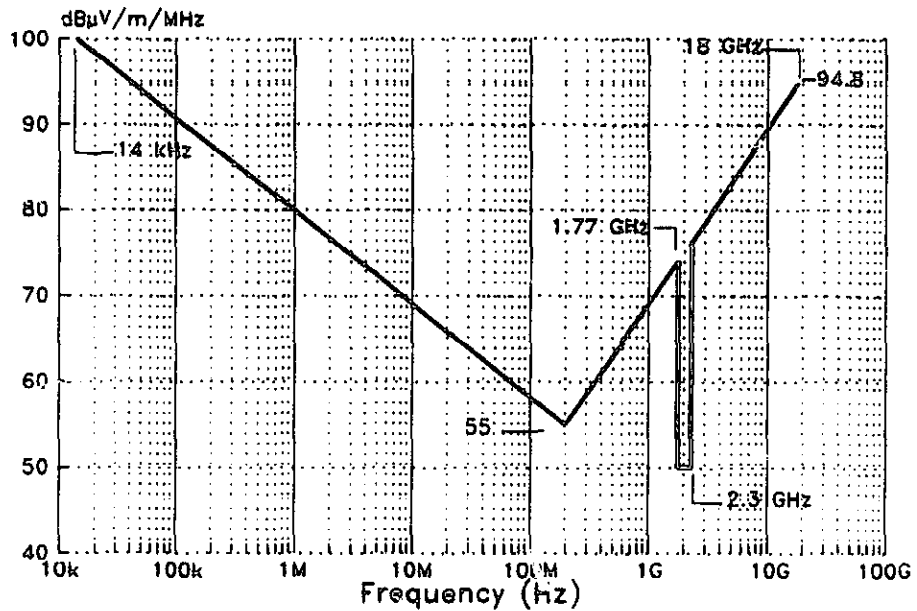


Figure 11. Limits, Unintentional BB Radiated E-Field Emissions

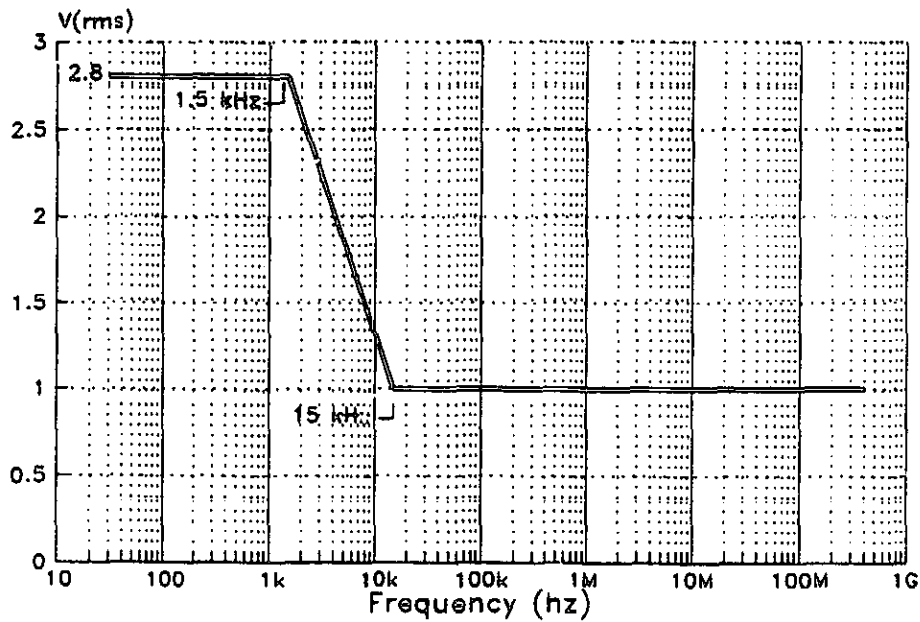


Figure 12. Test Levels for Conducted Susceptibility

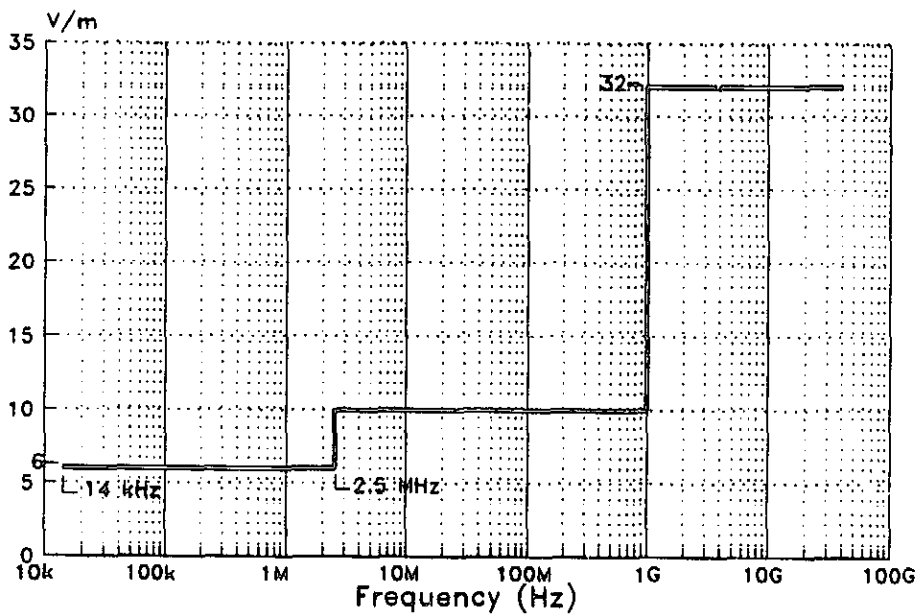


Figure 13. Test Levels for Radiated E-Field Susceptibility

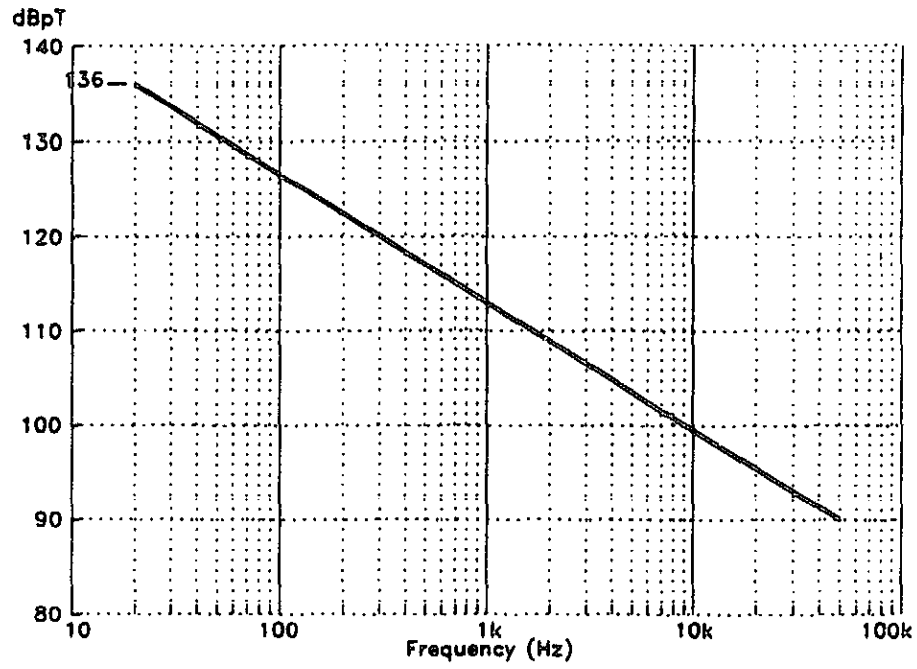


Figure 14. Test Levels for Radiated Magnetic Field Susceptibility

APPENDIX I - INSTRUMENT ENVIRONMENTAL TEST REQUIREMENTS

10.1 Protoflight (Qualification) Level Environmental Test Requirements

The Instrument engineering model (if applicable) and protoflight model shall be subjected to the following qualification level environmental tests. Except where noted the units shall meet all specified performance criteria during these tests, and the units shall be operated and configured during these tests in a manner simulating actual operating conditions expected during the various flight stages. In this appendix the letter "G" is used to designate an acceleration equivalent to one Earth gravity (ie approximately 9.81 m/sec^2).

10.1.1 Low Level Sine Survey.

A low level sine survey ($\approx 0.25 \text{ G}$ peak) shall be conducted on the instrument to identify the primary resonant frequencies, and to demonstrate that the structural resonance requirements have been met. The instrument shall be in the launch configuration for this test.

10.1.2 Static Loads Test (Acceleration).

A static load test shall be performed where applicable load specified in Figure 10.1 is applied along the direction of the velocity, anti-sun and nadir axes. The test item shall be in the launch configuration and shall be mounted in the shaker as it would be in the spacecraft. The test method may be either static load, acceleration or sine burst. The sine burst is recommended as the most practicable. The frequency used to perform the sine burst test is a function of both the dynamic characteristics of the test item and the limitation of the shaker facility. Because the test is intended to impart a static load to the test item, the test frequency should be less than one-third the test item resonant frequency to avoid dynamic amplification during the test. Should it not be possible to perform the test using sine burst one of the other alternatives shall be used.

Figure 10.2 shows a typical sine burst waveform envelope. The waveform is sinusoidal with a ramp up to maximum level, several cycles at maximum level, and then a ramp down to zero. A typical frequency is 20 Hertz, and the number of cycles at the maximum level is usually 6 to 10. The specification of a sine burst test shall contain the following information:

- o Test Level (as determined from Figure 10.1).
- o Test Frequency (less than one-third the fundamental resonant frequency of the test item).
- o Test Duration (6 to 10 cycles at maximum level).

10.1.3 Vibration Test.

10.1.3.1 Random Vibration.

The Instrument shall be subjected to the following qualification level vibration test in each of three orthogonal reference axes. During these tests each instrument shall be in the planned launch configuration. Requirement is defined in Acceleration Spectral Density (ASD).

<u>Frequency (Hz)</u>	<u>ASD (G²/Hz)</u>
20 - 50	+6dB/Octave
50 - 800	See Table 10.1
800 - 2000	-6dB/Octave

<u>Instrument Mass (kg)</u>	<u>ASD (G²/Hz)</u>
<10	0.19
10 - 14	0.07 X (M+20)/(M+1)
14 - 23	0.16
23 - 34	0.16 X (22.7/M)
34 - 65	0.07 X (M+20)/(M+1)
>65	0.09

Table 10.1 Peak ASD by instrument mass

10.1.3.2 Sinusoidal Vibration.

The Instrument shall be subjected to the vibration qualification test levels specified below in each of the three orthogonal reference axes. One sweep from 5 Hertz to 100 Hertz and one sweep from 100 Hertz to 5 Hertz shall be conducted for each axis. The sweep rate shall be two octaves per minute. During these tests the instrument shall be in the launch configuration. The following drive level requirements apply.

<u>Frequency</u>	<u>Amplitude/Acceleration</u>
5 - 18 Hz	Displacement ± 11 mm.
18 - 100 Hz	15 G peak

For components with first resonant frequencies close to 100 Hz, the drive level shall be reduced, as required, to achieve an instrument response of +15 G peak. Reduction in drive level shall in no case be to levels less than 5 G peak.

10.1.4 Shock Test.

The Instrument shall be subjected to the shock qualification test levels specified in Figure 10.3. The spectrum, pulse, or complex transient, is to be applied in each of the three orthogonal reference axes. During these tests the instrument shall be in the launch configuration.

10.1.5 Acoustic Test Requirements.

The instrument as part of the spacecraft payload during its environmental test sequence, will be exposed in the launch configuration to the acoustic levels shown in Table 10.2. Instruments designed with large area/low mass components which are exposed to and could be affected by direct acoustic energy shall have such components acoustically tested.

10.1.6 Launch Phase Pressure Profile.

The instrument design shall be such that exposure in the launch configuration to the environment specified in Figure 10.4 (times a factor of 1.12 on the rate of change), shall have no effect on instrument performance or life. The instrument shall be pressure profile tested if analysis does not demonstrate that a positive margin, at loads equal to 2.0 times those induced by the maximum expected pressure differential during launch, exists.

10.1.7 EMI/EMC Test Requirements (ref.).

The instrument shall meet all performance requirements when subjected to the tests specified in paragraph 3.5 of this document.

10.1.8 Particle Radiation Test.

The Instrument shall meet all specified performance requirements for the life of the spacecraft while exposed to the radiation environment specified in paragraph 3.8. The instrument shall be subjected to a supplier developed, and GSFC approved test which demonstrates that the instrument can absorb the predicted integrated dose and the peak dose rates. In lieu of a particle radiation test, an analysis, demonstrating that instrument incorporates adequate protective measures against the charged particle environment, may be performed.

10.1.9 Magnetic Field Tests.

10.1.9.1 Instrument Magnetic Properties.

The instrument shall be subjected to a supplier developed, and GSFC approved test which demonstrates that the instrument conforms to the magnetic properties limits defined in paragraph 3.4.3. In lieu of instrument testing, an analysis and/or subassembly tests, clearly demonstrating that the instrument conforms to the specified requirements, may be performed.

10.1.9.2 Magnetic Susceptibility.

The instrument shall be subjected to a supplier developed, and GSFC approved test which demonstrates that the instrument will operate within specification when subjected to the magnetic environments defined in paragraphs 3.4.1 and 3.5.3.2b. In lieu of instrument testing, an analysis and/or subassembly tests, clearly demonstrating that the instrument meets its specified

performance requirements in the spacecraft magnetic environment, may be performed.

10.1.10 Thermal Balance Test.

The performance of the instrument thermal design shall be determined and the instrument thermal analytic model shall be validated by subjecting the instrument to thermal balance testing. The instrument shall be in its flight configuration, and shall be mounted as it will be on the spacecraft to a temperature controlled interface whose temperature can be varied between the upper and lower operating and non-operating temperature limits specified in paragraph 3.3.2 for the spacecraft instrument mounting surface. The pressure during the test shall be no greater than 1.0×10^{-3} pascals. As a minimum requirement the worst case hot and cold conditions, as determined from analysis, shall be simulated on the instrument in both the operating and non-operating configurations. The orbital thermal environment including expected transients shall be simulated by the use of lamps, heaters and cooled shields.

10.1.11 Thermal Vacuum test Requirements.

The instrument shall be subjected to a qualification thermal vacuum test. The pressure shall be maintained at no higher than 1.0×10^{-3} pascals and the temperature profile shall be as shown in Figure 10.5 or a GSFC approved variant thereof. The temperature extremes shall be those established as worst case orbital limits for normal operation extended by 10 degrees Celsius beyond each limit as measured at the instrument mounting interface. During temperature transitions the rate of change in temperature shall not exceed ten degrees Celsius per hour nor be less than 5 degrees Celsius per hour. Time spent at each plateau shall be sufficient to allow the instrument to reach temperature equilibrium and to provide for the execution of the required performance tests, however, it shall be no less than four hours. The instrument is required to operate and survive over this temperature range, but need not operate within specification when the mounting interface temperature is less than -5 degrees Celsius or greater than +30 degrees Celsius unless the thermal analysis indicates broader limits should be imposed.

Shut-down and Re-start - During thermal vacuum testing, instrument shutdown and restart in orbit shall be simulated. At the extreme thermal vacuum temperature plateaus the instrument shall be shut down and the temperature allowed to stabilize at which time the instrument shall be restarted. The instrument shall be monitored to assure that its temperature does not go below its survival limit during stabilization at the low temperature plateau.

Instrumentation - Thermal instrumentation shall be attached to the instrument in sufficient number and in locations to measure the maximum and minimum structural temperatures as well as temperature critical items and those required for calibration

purposes. The instrumentation shall not invalidate the nature of the environment being measured. Care must be taken to insure that internal gradients are not generated which could jeopardize the integrity of the tests or materially affect the thermal performance of the instrument.

Chamber preparation - Prior to the installation of the instrument for thermal vacuum tests, the test chamber shall be cleaned; clean test fixturing and cabling shall be installed; and a temperature controlled quartz crystal microbalance (QCM) installed. The chamber shall be baked out under hard vacuum with the chamber walls maintained at the warmest temperature that they are expected to reach during the instrument thermal vacuum tests. The QCM shall be operated at 15 megahertz and maintained at minus 20 degrees Celsius. The chamber bake out shall continue until the QCM accretion rate as indicated by its rate of change of frequency is less than 35 hertz per hour.

Contamination Monitoring - The level of contamination present during the instrument thermal vacuum test shall be monitored using a temperature controlled QCM operating at minus 20 degrees Celsius and a frequency of 15 megahertz, strategically located witness mirrors, and, a cold trap for subsequent chemical analysis. During the tests the QCM accretion rate as indicated by its rate of change of frequency must be less than 80 hertz per hour.

10.1.12 Direct Arc Discharge Test.

The instrument shall be subjected to a supplier developed and GSFC approved test which demonstrates that the instrument is not susceptible to arc discharges.

10.2 Environmental Acceptance Test Requirements.

The environmental acceptance test requirements for the flight units differ from the protoflight level requirements as follows:

- (1) Delete test requirements of paragraphs 10.1.6, 10.1.8, 10.1.9, 10.1.10, and 10.1.12;
- (2) In test requirements of paragraph 10.1.2 reduce the test level by 1.25;
- (3) In test requirements of paragraph 10.1.3.1 reduce levels by a factor of 2 for the PSD and a factor of 1.4 for any derived RMS specified levels;
- (4) In test requirements of paragraph 10.1.3.2 reduce the levels by a factor of 1.25 and change the sweep rate to 4 octaves per minute;
- (5) In test requirements of paragraph 10.1.4 use the lower curve in Figure 10.3;

(6) In test requirements of paragraph 10.1.5 use the acceptance levels of Table 10.2

(7) In test requirements of paragraph 10.1.11 reduce the temperature extremes to the predicted worst case orbital levels.

Tables and Figures

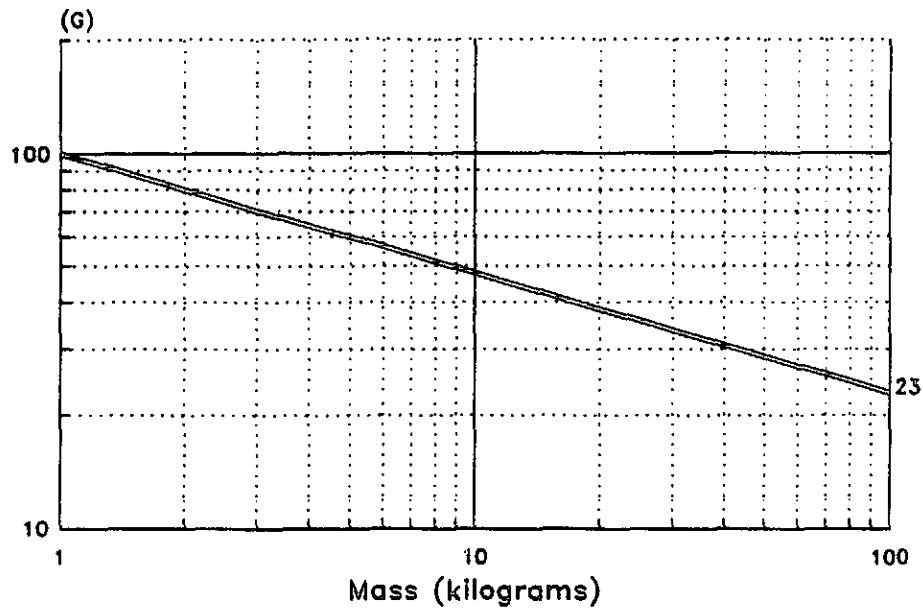


Figure 10.1 Linear Acceleration Requirements

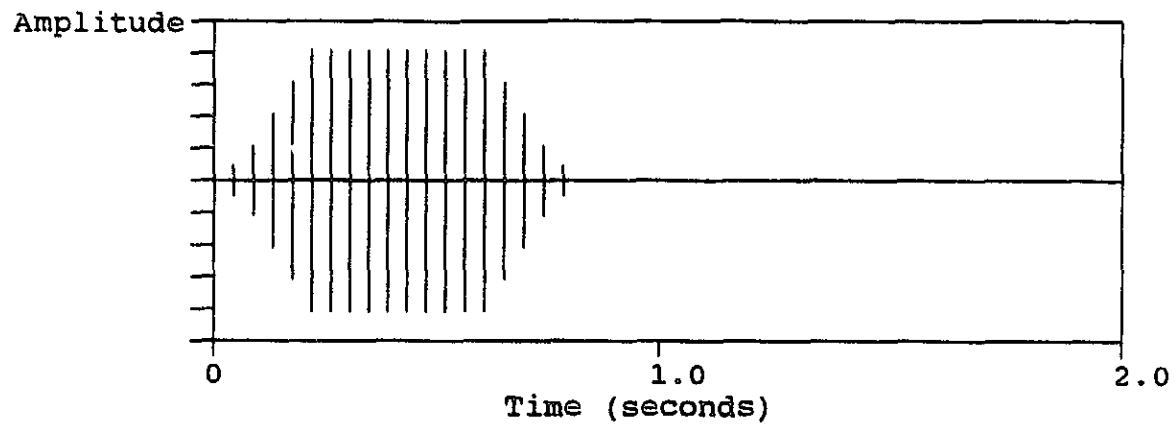


Figure 10.2 Sine Burst Waveform Envelope (typ)

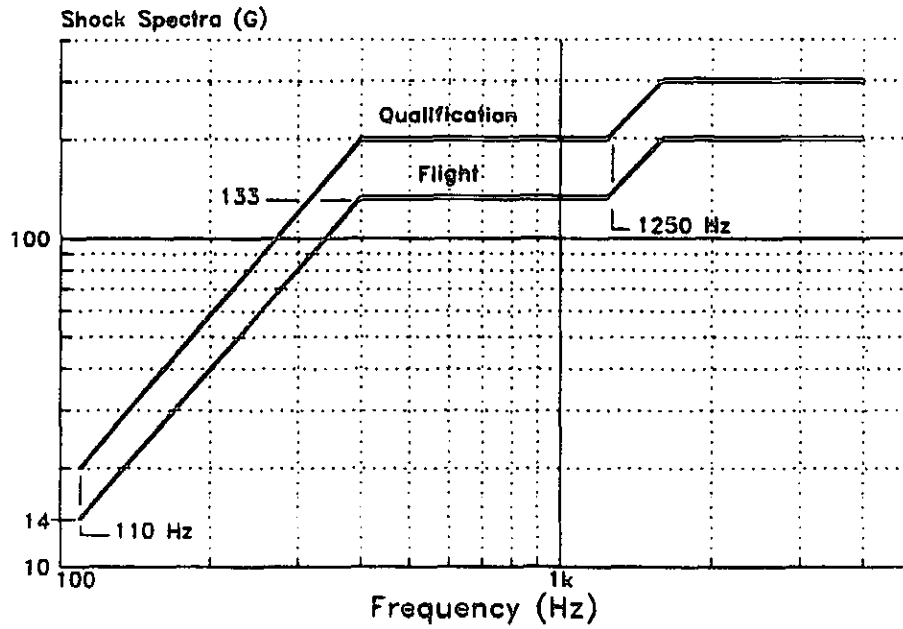


Figure 10.3 Qualification and Flight Level Shock Spectra

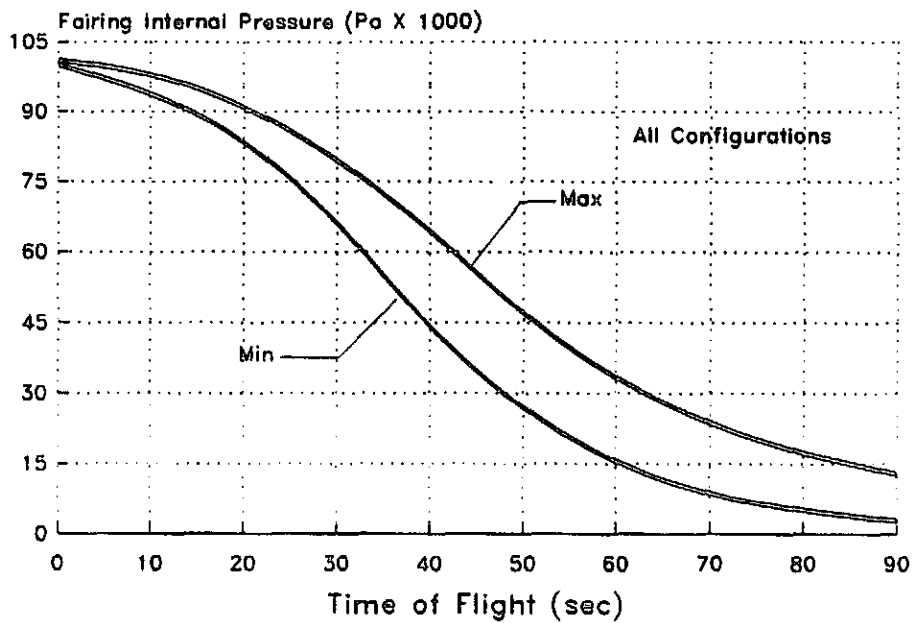
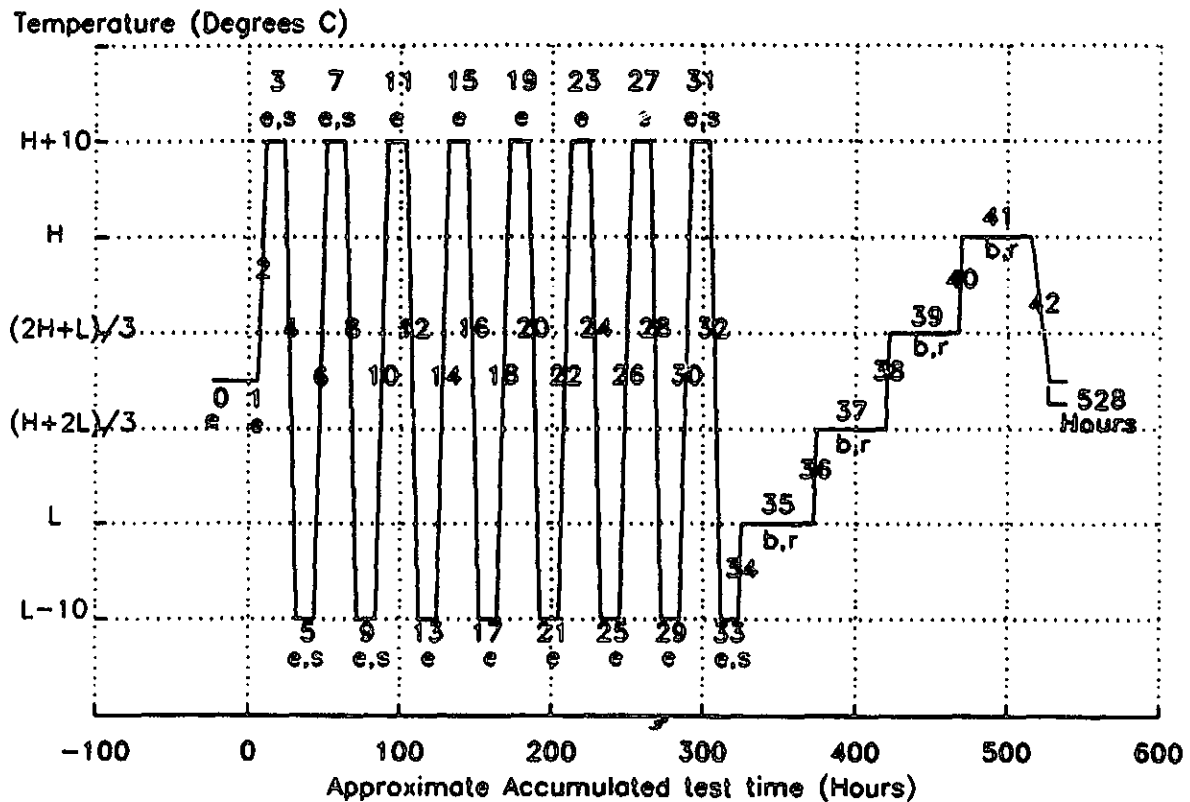


Figure 10.4 Launch Phase Pressure Profile



Notes: H = High Operating Temperature Limit

L = Low Operating Temperature Limit

Numbers on face of chart designate test operations zones (See Table 10.3)

Operations Codes are: e = electrical functional
s = Shutdown/Soak/Restart
b = Stabilize bake
r = Radiometric Calibration

Table 10.3 describes operations to be conducted during each test zone.

Figure 10.5 Thermal-Vac Qualification Level Test Profile

Table 10.2 Acoustic Test Levels

One-Third Octave Center Frequency (Hertz)	Noise Level (dB) re: 0.00002 Pa	
	Qualification	Acceptance
25	-	-
32	126	123
40	130	127
50	133	130
63	135	132
80	136	133
100	136.5	133.5
125	137	134
160	137	134
200	137	134
250	137.5	134.5
315	138.5	135.5
400	138.5	135.5
500	136	133
630	131	128
800	126	123
1000	122.5	119.5
1250	120	117
1600	118.5	115.5
2000	117.5	114.5
2500	115.5	112.5
3150	112	109
4000	110.5	107.5
5000	110	107
6300	109.5	106.5
8000	109.5	106.5
10000	109.5	106.5
Overall	147.5	144.5

Table 10.3 Thermal Vacuum Test Sequence

Zone	Time (hours)	Activity
0	--	Prepumpdown Electrical Functional
1	6	Post Pumpdown Electrical Functional
2	6	Orbit Simulation
3	12	Electrical Functional and Shutdown/Restart
4	8	Orbit Simulation
5	12	Electrical Functional and Shutdown/Restart
6	8	Orbit Simulation
7	12	Electrical Functional and Shutdown/Restart
8	8	Orbit Simulation
9	12	Electrical Functional and Shutdown/Restart
10	8	Orbit Simulation
11	12	Electrical Functional
12	8	Orbit Simulation
13	12	Electrical Functional
14	8	Orbit Simulation
15	12	Electrical Functional
16	8	Orbit Simulation
17	12	Electrical Functional
18	8	Orbit Simulation
19	12	Electrical Functional
20	8	Orbit Simulation
21	12	Electrical Functional
22	8	Orbit Simulation
23	12	Electrical Functional
24	8	Orbit Simulation
25	12	Electrical Functional
26	8	Orbit Simulation
27	12	Electrical Functional
28	8	Orbit Simulation
29	12	Electrical Functional
30	8	Orbit Simulation
31	12	Electrical Functional and Shutdown/Restart
32	8	Orbit Simulation
33	12	Electrical Functional and Shutdown/Restart
34	2	Orbit Simulation
35	46	Stabilize and Radiometric Calibration
36	2	Orbit Simulation
37	46	Stabilize and Radiometric Calibration
38	2	Orbit Simulation
39	46	Stabilize and Radiometric Calibration
40	2	Orbit Simulation
41	46	Stabilize and Radiometric Calibration
42	<u>12</u>	Power down and bleed-up the Chamber
Total	528	